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THE EARTH OBSERVATORY SATELLITE (EOS)

A SYSTEM CONCEPT

EOS PROJECT OFFICE
GODDARD SPACE FLIGHT CENTER
GREENBELT, MARYLAND 20771

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NOTE

This document outlines some of the background and system considerations which have led to a preliminary concept of the Earth Observatory Satellite (EOS) System. This study is a part of the effort leading to the identification and evaluation of different options for future earth observation systems. This report is for general information only and should not be construed as a commitment by NASA management to either the configuration or mission options identified herein.

EARTH OBSERVATORY SATELLITE (EOS)
A SYSTEMS CONCEPT

1.0 INTRODUCTION

Since 1970, NASA planning has identified the need for an Earth Observatory Satellite (EOS) as the system for Applications Missions in low earth orbit. Overall objectives of the EOS Program are to develop economical, multipurpuse, modular spacecraft systems to carry observing techniques of the late 1970's and the 1980's for:

a. Earth and ocean surveys.
b. Pollution detection and monitoring.
c. Weather and climate prediction.

EOS-A Mission Objectives are:

1. Develop sensor and other spacecraft systems to acquire spectral measurements and images suitable for generating thematic maps of the earth's surface.

2. Operate these systems to generate a data base from which land use information such as crop or timber acreages or volumes, courses and amounts of actual or potential water run-off and the nature and extent of stresses on the environment will be extracted.

3. Demonstrate the application of this extracted information to the management of resources such as food and water, the assessment and prediction of hazards such as floods, and the planning and regulation of land use such as strip mining and urbanization.

EOS-A is planned as the first of a series of opportunities for flight test of earth observation instrument developments, and research in applications of the data.

1.1 TECHNICAL PLAN

To accomplish mission objectives it will be necessary to:

1. Develop space-borne sensors for the measurement of parameters, as required by earth observations discipline objectives, with increased performance and in new spectral regions not achievable by present sensors.

2. Evolve spacecraft systems and subsystems which will permit earth observations with greater accuracy, coverage, spatial resolution and continuity than existing systems by avoiding spacecraft constraints on sensor performance.

3. Develop improved information processing, extraction, display and distribution systems so that the applicability of the observations may be enhanced.

4. Achieve the objectives above with sufficient economy and flexibility to permit the operational use of any hardware or other system component with little or no redevelopment.

5. Use the space transportation system's resupply and retrieval capability to sustain and refresh this remote sensing capability through the 1980's, thereby providing an efficient means for demonstrating the viability of improvements prior to committing to operational use.

A key goal for the development of an advanced remote sensing system is to increase the proportion of project funds allocated to payload, as compared to the spacecraft systems which support the payload. These considerations lead to a modular observatory system, the space borne elements of which are compatible with the Delta, Titan, and shuttle launch vehicles. The scope of any particular mission can be sized to fit funding and launch vehicle constraints without impacting the design of individual modules. The major system engineering and technology development tasks involve the instruments and their data management systems. Therefore they will receive the initial emphasis along with their system integration requirements. Spacecraft subsystem fabrication will be delayed until the implementation schedule requires initiation. Sufficient hardware will be procured for a backup mission. The observatory's compatibility with the space shuttle's launch, refurbishment, resupply and retrieval modes will be verified by an early shuttle flight from ETR.

1.2 MANAGEMENT PHILOSOPHY

The first decade in space typically contained projects where the spacecraft developments were more complex than their instrument payloads. The reverse is true for EOS. Existing spacecraft hardware can meet EOS requirements whereas the instruments and their data management system are very complex developments. The following factors have shaped the EOS management philosophy and approach.

1. Assurance that the major instruments are feasible, prior to commitment to System Execution, will be satisfied by competitive breadboard developments for the Thematic Mapper, and extensive definition studies for both the Thematic Mapper and High Resolution Pointable Imager.

2. The objective of developing EOS with sufficient economy to meet follow-on operational cost constraints makes overall systems engineering mandatory. Therefore, system engineering and cost optimization will be the major task for the system definition study contractors.

3. All EOS instruments are complex but some are mission drivers while others have little impact on the rest of the system. In order to hold the flight system contractor responsible for the overall system output, the first class of instruments and their associated data management systems will be placed under the execution phase system contract. This simplified interfaces and should minimize change orders. The second class of instruments will continue to be provided as GFE.

4. The achievement of a low-cost program will require a joint Government-industry effort focused on this goal. The System Definition Contractors will be encouraged to propose new management approaches that will produce a lower cost program without impacting reliability.

5. The adaptability of the modular spacecraft subsystems allows them to meet mission requirements beyond EOS. Contracts will be structured to preserve the identity of these modules. Management plans will be established to provide options for commonality at either the system or subsystem level.

6. The shuttle-based special purpose manipulator is planned as GFE to the EOS System Contractor. This plan will accommodate the potential Canadian involvement and to provide the option for use on missions beyond EOS-A.

2.0 RELATED STUDIES AND ACTIVITIES

Detailed planning for the EOS Missions was initiated at GSFC in June 1970. At that time the Earth Observations R&D Program consisted of two major elements, the Nimbus Meteorological satellite series, authorized thru Nimbus F in 1974, and the Earth Resources Technology Satellite Program authorized thru ERTS-B in 1973. Two Working Groups at GSFC had addressed the question of appropriate follow-on to these missions.[1,2] Both Working Groups identified a series of priority problems or opportunities addressable by remote sensing from space which were not addressed in the Nimbus and ERTS Programs as then defined. Both working groups were in agreement that a follow-on Earth Observations Program was needed. They also found large overlap of Meteorological and Earth Resources Missions and agreed that consideration should be given to combining the Nimbus and ERTS Missions into an Earth Observation Mission.

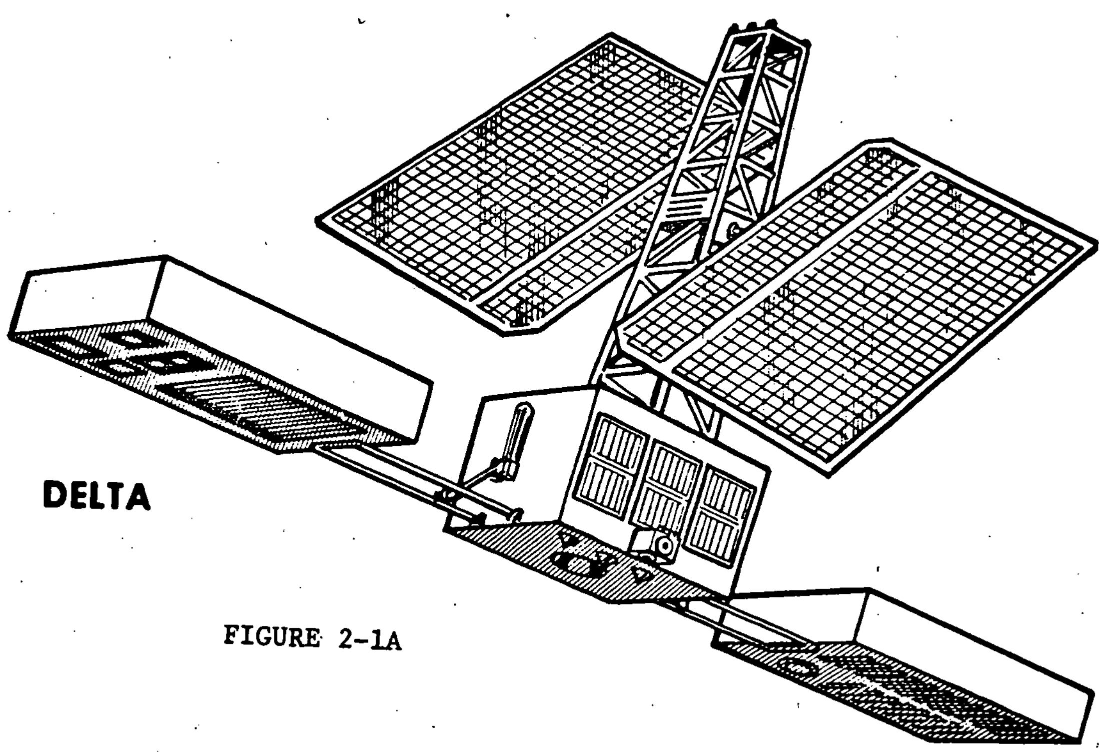
The GSFC study was accomplished in-house by a group representing discipline areas, instrument design, spacecraft systems, and data management systems. Strong support in the areas of user agency needs and instrument measurement parameters was provided by a group of consultants from user agency groups, many of whom actively participated not only in the definition of the Earth Observations Missions but also in the preparation of the final report.

The EOS Definition Phase Report[3] was completed in August 1971. In the course of the study an extensive effort was devoted to the data needs of the earth observation disciplines and to the development status of instruments which could meet these needs. A number of instruments were identified which could be available for flight late in the 1970's; others were identified which would require a longer development cycle. Preliminary design concepts of key instruments were prepared. Some of these instruments were found to weigh from 350 to over 500 pounds in contrast to the heaviest single instrument flown in the ERTS/Nimbus Program which weighed 204 pounds. Combinations of instruments were developed which provide related data and for which the combined effectiveness is greater than the sum of individual parts.

[1]Nimbus G and H, Mission Requirements and Concepts (Pre-Phase A) Study. Meteorology Program Working Group, Goddard Space Flight Center, December 20, 1969.

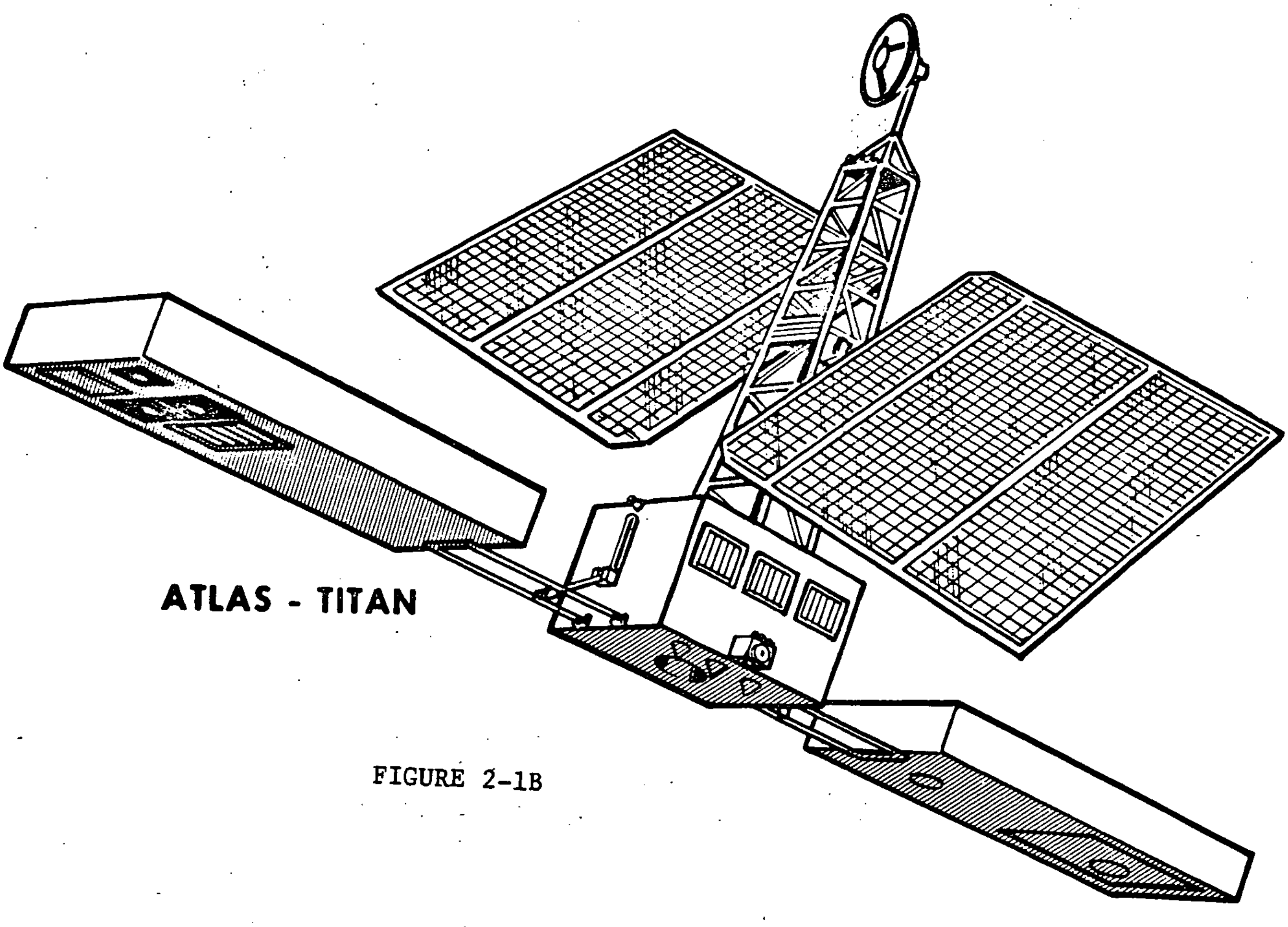
[2]Outline of Follow-on Missions to ERTS A and B by GSFC Earth Resources Working Group, April 16, 1970.

[3]Earth Observatory Satellite (EOS) Definition Phase Report, X-401-72-332, GSFC, August 1971.



DELTA

FIGURE 2-1A



ATLAS - TITAN

FIGURE 2-1B

ORIGINAL Φ 'A' EOS CONFIGURATION

The study examined the relationship between instrument characteristics and the spacecraft performance parameters which would support these instruments. It was found that no known existing spacecraft would be adequate for the EOS Missions.

For early flights it was concluded that a Delta launched spacecraft as shown in Figure 2-1A would be adequate to initial mission goals provided additional lifting capability was provided, but that the course of instrument development was such that a larger spacecraft of the Atlas-Titan class would very likely be required before 1980 as shown in Figure 2-1B.

Upon completion of the GSFC study, its principal results were presented to the Earth Resources Survey Program Review Committee (ERSPRC) and to the Meteorological Satellite Program Review Board (MSPRB). At the recommendation of these boards NASA established the EOS Mission Review Group (EOSMRG) "to finalize, based on discipline objectives and user agency requirements the mission objectives and sensor performance specifications for the first two missions (EOS-A and -B) and to formulate requirements for the management of the EOS data."

Membership in the EOSMRG was drawn from NASA, DOD, USDA, USDI, USACE, and EPA. The Group met at intervals from August to November 1971. The report[4] modified the instrument recommendations of the Phase A report (with a recommendation for decreasing the proposed 66 meter IFOV of the Thematic Mapper to less than 50 meters with the same swath width (100 n.m.). In addition the EOSMRG had a number of recommendations which could lead to payload additions. These recommendations included:

High Resolution Pointable Imagers

Framing Camera Mappers

Atmospheric Pollution Sensors (Particulates)

Atmospheric Temperature Sounders

Adbedo-Radiation Budget Instruments

Data Collection and Location Systems

A concerted effort to develop an imaging radar system for flight at the earliest opportunity.

[4]Earth Observatory Satellite Mission Review Group (EOSMRG) Final Report, GSFC, November 1971, X-401-72-333.

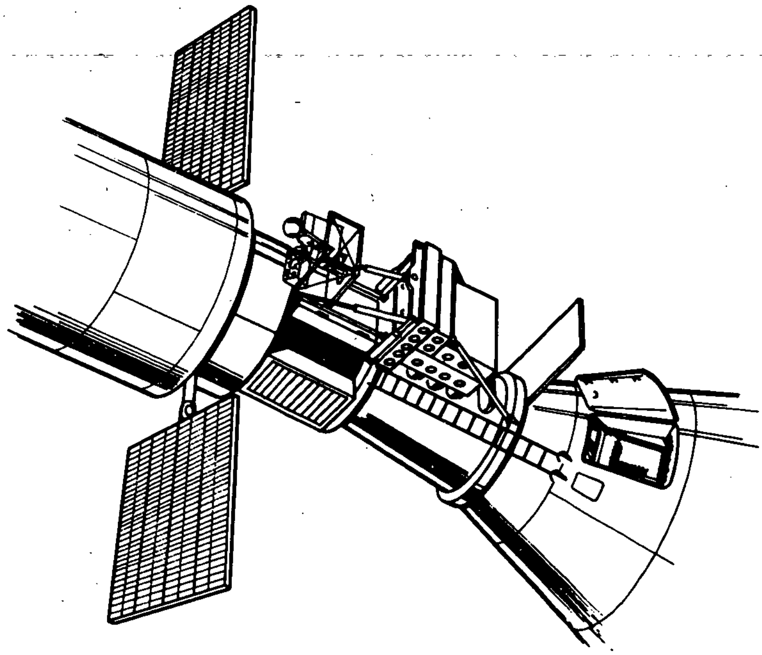
The EOSMRG studied a number of instrument combinations for the initial EOS flights. These payloads covered a number of options including four discipline oriented, and two inter-disciplinary missions. In the data management area the Group highlighted the need for realtime processing of selected data (as an example data required for disaster assessment or resource modeling). The group also recommended a cooperative planning effort with the user agencies leading to common-use analysis facilities. The EOSMRG report provided a firm foundation for processing with system planning.

With the establishment in FY 1971 of UPN 613 for the EOS Definition Phase, a number of developments were undertaken which with the SR&T program were designed to verify instrument performance parameters, establish the feasibility of the space-craft on-board data processing systems, and to examine some of the relationships between the ground trace of certain instrument scanning patterns and the ground data processing required to produce imagery. In view of the projected NASA level budget, and in view of the planned space shuttle as a potential new factor in permitting a continuous observing capability by means of on-orbit resupply, intensive studies were undertaken of modular spacecraft system approaches which could exploit the shuttle when it becomes available, and which could be provided at low cost for launch on expendable boosters in the pre-shuttle transition era for both the EOS and other missions.

SYSTEM CONFIGURATION BACKGROUND

In 1967 the OAO Project initiated studies of advanced designs which eventually evolved into the LST. The initial efforts recognized the potential of the Centaur launch vehicle for greater payloads, and the growth capability of the basic OAO spacecraft to accommodate larger telescopes and instruments. A NASA Headquarters initiated study of optics and instrumentation paralleled the Goddard Space Flight Center (GSFC)/Grumman Aerospace Corporation (GAC) observatory configuration study. In the ASTRA work it was assumed that a manned maintenance capability would exist, and the extremes of simply resupplying expendables to direct involvement by man in replacing black boxes were considered.

It became apparent in this early work that there were serious drawbacks to in-orbit maintenance if man is directly involved in black box replacement. It appeared that it would be more economical to continue along the fully automated path than to fund the man rating of an observatory and to bear even a portion of the cost of an associated space station. In the period following the ASTRA study, techniques were developed for maintaining spacecraft in orbit without direct and hence costly involvement by man. In this concept, the observatories would be modularized at the subsystem level for in-orbit replacement by means of an astronaut controlled mechanism as shown in Figure 2-2.

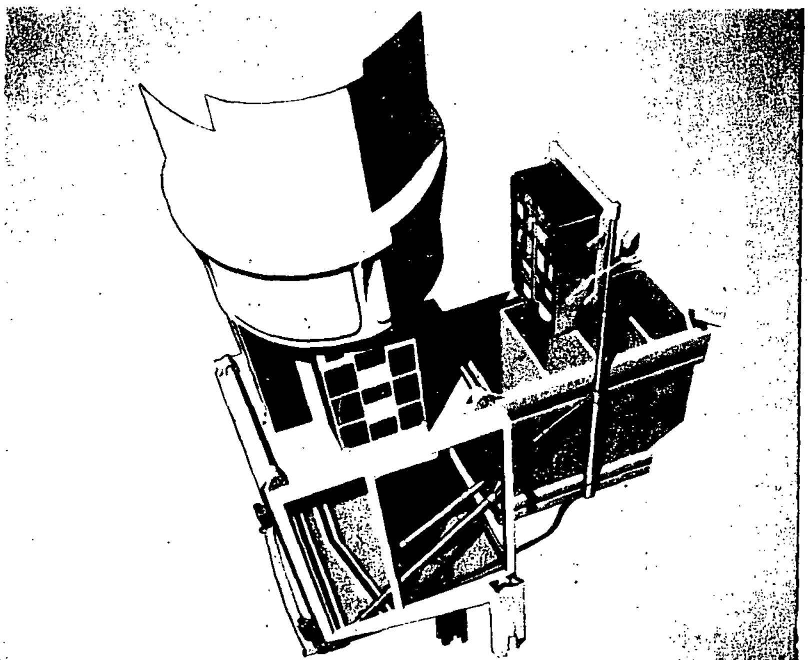


RESUPPLY CONCEPT

FIGURE 2-2

In this approach, manned maintenance was associated with a space station, and an Apollo type Command Service Module (CSM) was assumed for sortie missions between the station and the free flying observatories. The mechanism for servicing the observatories and the space modules would be attached to the exterior of the CSM. Since this program would have required a substantial portion of the space station resources, it appeared to be too expensive on a logistics cost basis.

In 1970, the prospect of launching and servicing a spacecraft with a space shuttle triggered a reassessment of the potential of in-orbit maintenance. The shuttle need not be dedicated to a handful of observatories, as was the case with the space station, and therefore the cost of maintenance could be moderate. The largely automated means for resupply eliminated the cost of man rating the observatory as shown in Figure 2-3.



RESUPPLY ACTIVITY

Figure 2-3

Further cost benefits of this approach became apparent when it was shown that in-orbit maintenance could reduce the number of observatories required for a scientific program. The space shuttle could be used to update experiment instruments as well as to perform maintenance functions. In the event of a nonrepairable failure in orbit, the entire spacecraft could be retrieved. Since the spacecraft could be repaired, less expensive hardware and testing can also be used. The results of a study to quantify these benefits are given in "OAO/LST Shuttle Economics Study" by Grumman Aerospace Corporation, October 1970. During the course of the LST studies, it became apparent that the basic spacecraft modules which were designed for LST could be readily adapted for alternate scientific or earth viewing missions.

In 1972, during the course of the EOS study, engineering models were designed, fabricated, and tested to prove out the mechanical feasibility of subsystem modular replacement. Tests conducted both at GSFC and JSC have conclusively demonstrated the feasibility of modular subsystem replacement. During the EOS Study and technology demonstration efforts, the concept of subsystem modularity made possible a new management approach that produced the single most significant advantage of standard subsystem modules - lower costs.

In FY 72, 73, and 74 SR&T activities were carried out to develop a better understanding of shuttle interfaces as well as the definition and development of the technology needed to perform EOS/Shuttle docking and resupply activities. Included in this area was the development and vibration-proof test of a full scale modular spacecraft. One of the principle objectives of these activities was to develop a better understanding of the performance, the resupply mechanisms, the nature of the interfaces and their impact on the environmental levels of the modular subsystems and the S/C system.

The supporting mechanical system activities being carried out in support of the project are as follows:

1. Shuttle/EOS interface study.

2. Low cost demonstration model development.

3. Resupply technology tests at JSC.

4. Canadian special purpose manipulator study.

3.0 TECHNICAL SUMMARY

3.1 MISSION CHARACTERISTICS

LAND RESOURCE MANAGEMENT - The initial mission in the EOS series is a Land Resource Management Instrument Development Mission. This mission will develop advanced instruments which can provide multispectral imagery of the land surface of the earth at significantly improved spatial, spectral, and temporal resolutions than is available from the Earth Resources Technology Satellite or from the projected Department of the Interior Earth Resource Survey Operational System, and thus will permit studies of the direction in which the operational land use inventory, and earth resource management programs should proceed. Initial flight test of the instruments and applications research with the data can be made possible by launch of the projected EOS-A in 1978.

OCEANOGRAPHY-METEOROLOGY - A second EOS mission which could be launched in 1979 is the Oceanography-Meteorology Instrument Development Mission. This mission is addressed to the priority problems of Oceanography and Meteorology, especially those associated with an improved data base for long range weather forecasting, and for ocean resource modeling. Initiation of instrument development in FY 76 will permit launch in 1979. Many of the areas for research in this time period are inter-disciplinary in that they involve the disciplines of both meteorology and oceanography.

Five specific areas requiring further research are:

A. Sea Surface Pnenomena (temperature, roughness, color).

B. Structure and Phenomena of the Atmosphere above 30 km (temperature, composition.

C. Cloud Structure and Composition.

D. Spectral, Spatial, and Temporal Characteristics of Selected Earth Surface Features (soil moisture, ice and snow cover).

E. Interactions Between Different Levels in the Earth Atmosphere System.

In addition, instruments initiated in the Advanced Applications Flight Experiments (AAFE) program relevant to any of the earth observations disciplines especially pollution monitoring may be selected for a 1979 flight opportunity.

ALL WEATHER OBSERVATORY - The global measurements of atmospheric structure made possible by remote sensing techniques from satellites are a major portion of the data base for the numerical models used for forecasting. Most techniques used for structure determination depend on measurements in the infra-red and are thus applicable only to cloud free situations. The achievement of an all weather capability for both atmospheric structure determination and for surface observation has been termed a major goal of remote sensing from space. An All Weather Observatory in 1981 is a timely goal for the third EOS Mission. There will have been time to assess the utility of the microwave radiometers on Nimbus G and EOS-B, the synthetic aperture radar proposed for EOS-A and the Radar Scatterometer for Skylab A. Advanced versions of these instruments, or reflights as appropriate, coupled with new developments will be feasible. Detailed definition of the payload will be undertaken in connection with the FY 78 budget cycle.

SHUTTLE RESUPPLY DEMONSTRATION TEST FLIGHT - In order to verify the observatory's compatability with the shuttle capability for launch, resupply and retrieval, an early shuttle flight is planned from ETR using the EOS backup or engineering model hardware. The objective of the flight shall be a final "shake down" of the combined shuttle, the resuppliable observatory and the flight support system which interfaces the standard shuttle with the unique EOS payload system.

In order to factor any necessary design changes into the flight support system or the shuttle prior to the first resupply of the EOS-A mission, it is highly desirable that this demonstration "shake down" flight be flown among the first six checkout flights of the shuttle. These flights will be flown from ETR at low inclination for the purpose of checking out the shuttle systems prior to operational status.

3.2 SYSTEM

To achieve truly low cost system design, a modular "building block" approach has been adopted. The heart of this concept lies in the ability to use a set of nonoptimized subsystems in such a way that a variety of missions can be supported. By standardizing the mechanical configurations and electrical interfaces of the subsystem modules, and by designing each of them to be structurally and thermally independent entities, it is possible to cluster these building blocks or modules about an instrument system so as to adequately perform the mission without the need for subsystem redevelopments for each mission.

This system concept offers the following capabilities:

1. The ability to launch and orbit the observatory by either the Delta, the Titan or the space shuttle.

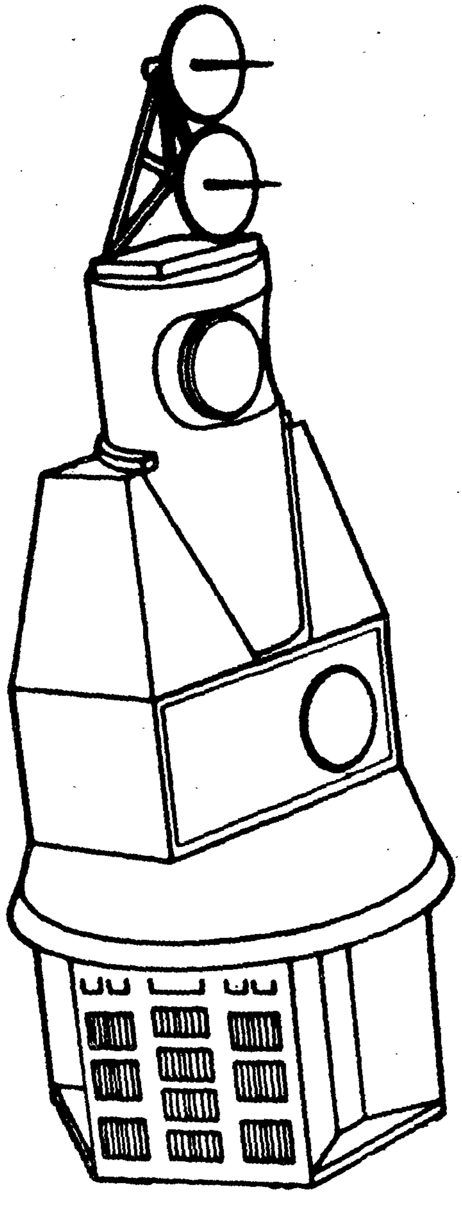
2. The ability to completely reconfigure the spacecraft subsystem for different payloads and launch vehicles.

3. The ability to perform in orbit resupply and/or emergency retrieval of the observatory.

For the purpose of description the configuration of the EOS Observatory consists of three basic sections or assemblies (Figure 3-1): As shown in a Delta configuration.

1. An instrument payload assembly whose configuration is completely mission peculiar and independent of the subsystem structural and thermal design.

2. A subsystem structural assembly whose function it is to contain the subsystem modules and to ultimately provide the structural interface for shuttle resupply activities.



EOS/DELTA CONFIGURATION

FIGURE 3-1

3. A transition ring assembly which functions as the structural interface between the subsystem assembly and mission peculiar instrument payload assembly. Furthermore it is the function of the transition ring assembly to transfer all launch loads from the instrument assembly and the subsystem assembly directly to the launch vehicle. In this manner both the subsystems and instrument assemblies are designed to carry only their own loads.

CONFIGURATION POTENTIALS

The configuration flexibility of the basic subsystem approach is illustrated by Figure 3-2 which depicts a Titan launch vehicle configuration in which these same standard subsystems can be packaged:

1. To house multiple resuppliable instruments within a 10 foot diameter Titan shroud,

2. To house a full up NASA R&D mission instrument complement within the shuttle. The Delta configuration is currently shuttle retrievable but not resuppliable wereas the latter two offer both resupply and retrieval capabilities.

SYSTEM THERMAL CONTROL CONCEPT

The objectives of the overall system thermal control concept are as follows:

1. Provide flexibility for accommodating different orbit and mission configurations.

2. Design for thermal independence of individual subsystem and instrument modules.

3. Maximize the thermal isolation of subsystems modules from instrument modules.

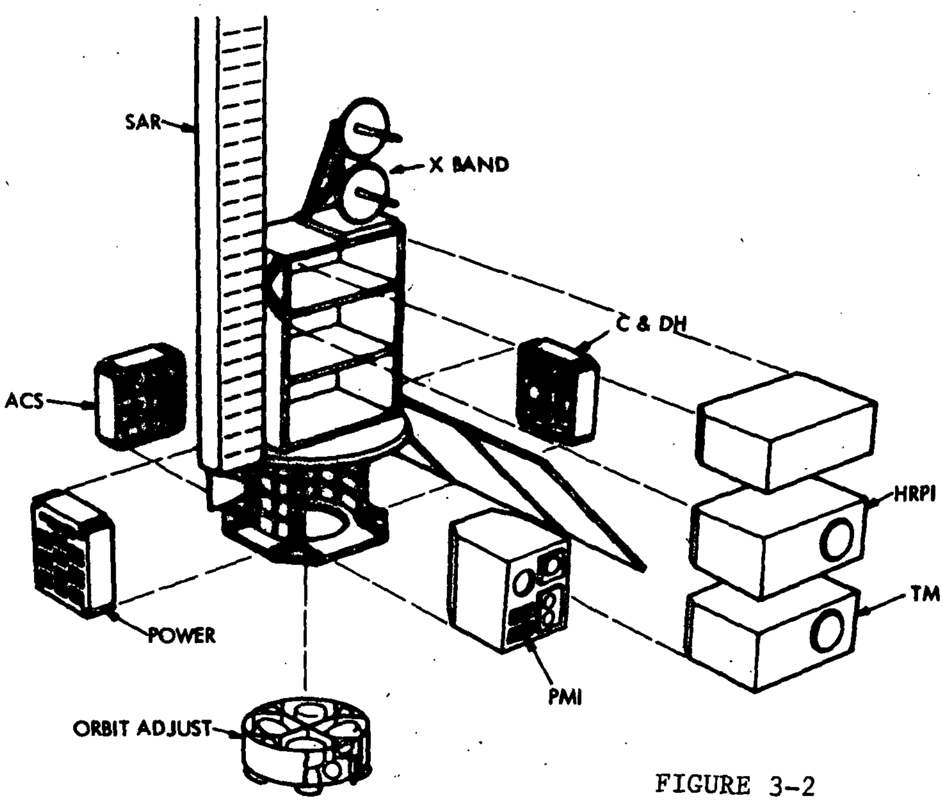


FIGURE 3-2

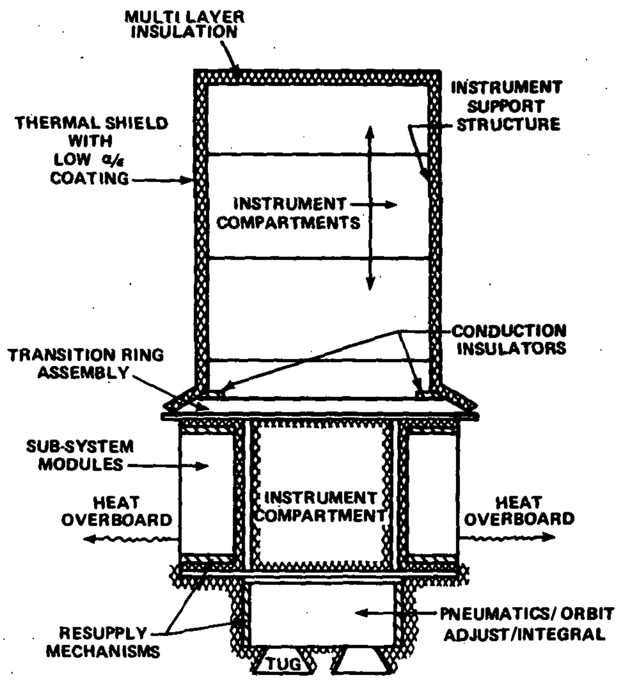
EOS MODULARITY

4. The selection of nominal (70°F) operating temperature for each module and the minimization of the excursions about that set point (≠20°F) so as to provide minimum stress to electrical components.

5. The potential elimination of system level thermal vacuum test since the module thermal design is relatively insensitive to its placement in the system.

In the EOS modular concept many of the above objectives are achieved through the thermal design of each module such that it maintains complete thermal independence from adjacent modules and supporting structure. This is to be accomplished by having an insulated interface between each subsystem module and instrument module as shown in Figure 3-3. If necessary, heater power internal to each module could be used to hold a predetermined temperature level. Each subsystem has its own radiative dump capability to space, and this external radiating surface will be an integral part of the thermal design of each module. The subsystem module support structure will be bounded by insulated interfaces that will normally separate it from the temperature-controlled modules. Therefore, its heater power requirements, during normal operation, will be small, and individual heaters will be located so as to minimize the thermal gradients in the structure.

In this independent thermal control concept, thermal exchange between modules and the system is drastically curtailed. Each module can therefore be realistically tested by itself to its in-orbit operational environment, thereby eliminating the need for a system-level thermal-vacuum test to confirm thermal design.



TYPICAL SYSTEM THERMAL CONTROL CONCEPT.

FIGURE 3-3

3.2.1 Instruments

3.2.1.1 Thematic Mapper

For the purpose of delineating land-use "themes", the EOSMRG in their report of November 1971* recommended as the primary instrument a moderate resolution multi-spectral scanner with an instantaneous field of view (IFOV) of no greater than 50 meters. A candidate instrument was under study at that time with an IFOV of 66 meters; subsequent studies indicate an IFOV of 30 meters at 900 kilometers altitude to be feasible.

Significant parameters of the EOS-A/LRM Thematic Mapper are tabulated below:

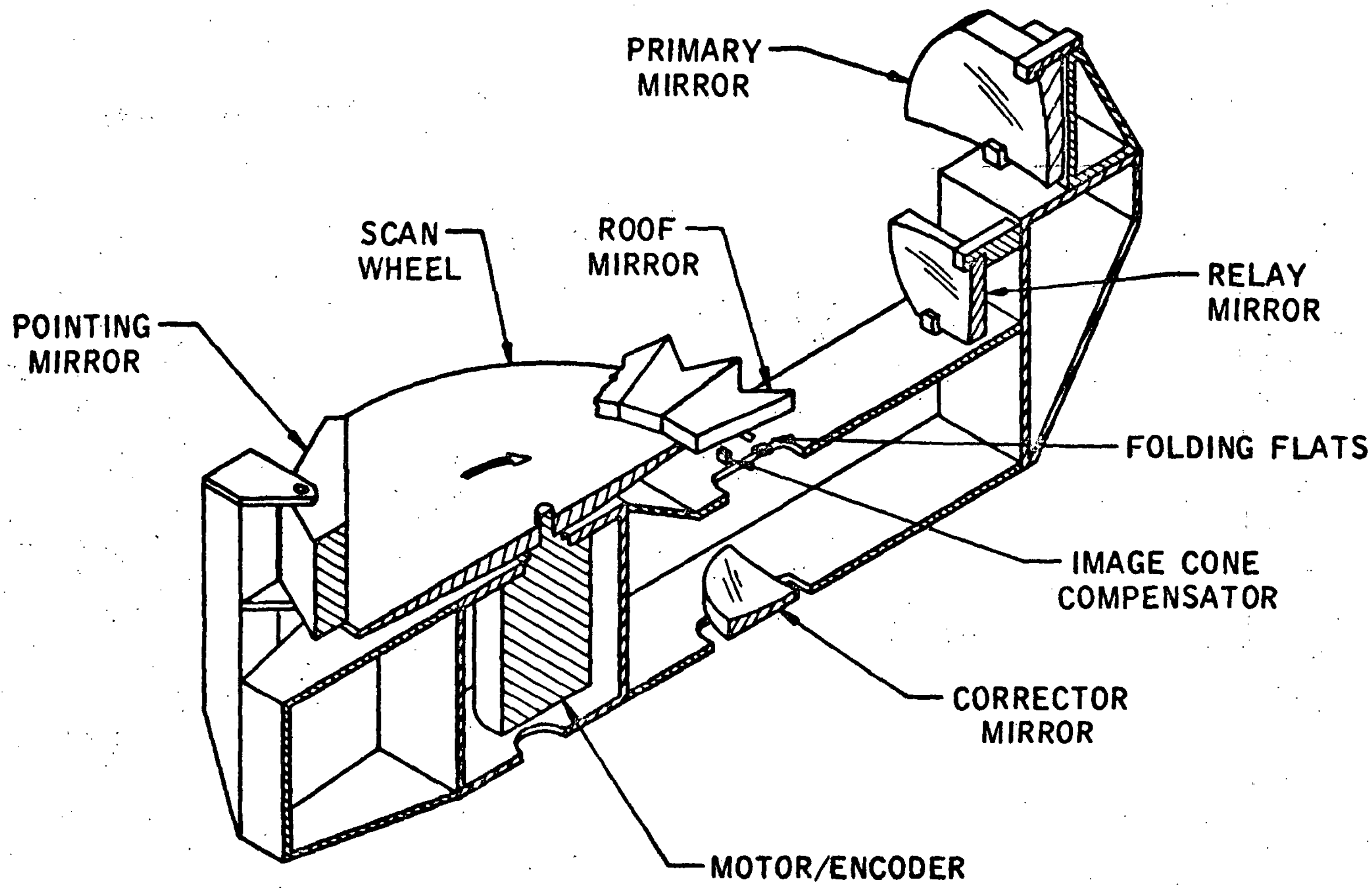
Number of Spectral Bands:	6 or 7 (TBD)
IFOV	30 meters @914-km Altitude
Swath Width	185.2 kilometers
Size	Cylinder, 0.914 meter diameter x 2.13 meters long (36 inches x 84 inches)
Weight	272.7 kilograms (600 pounds)
Power (exclusive of heaters)	100 watts
Real-time output bit-rate	120 Mbps including overhead

*Terrestrial Resources Panel Report and Environmental Quality/ Ecology Panel Report, Earth Observatory Satellite Mission Review Group Final Report (November 1971).

Feasibility and system cost effectiveness of three different approaches to scanning will be determined under AAFE/EOS contracts. Point designs of these instruments have been completed and are illustrated in Figures 3-4A, 3-4B, and 3-4C.

3.2.1.2 High Resolution Pointable Imager (HRPI)

The High Resolution Pointable Imager (HRPI) was originally recommended by the EOSMRG as a high-resolution adjunct to a medium-resolution Thematic Mapper; its role was further developed by the December 1972 KSC Conference on Imaging for Earth Resources as an experimental tool for determining the efficiency of long-term high-resolution sampled data obtained on a regular programmed basis which could reduce the dependence on aircraft data. In addition to its normal use in this mode, the HRPI can be used for ad-hoc temporary purposes such as flood or hurricane damage asssessment.



THEMATIC MAPPER - IMAGE PLANE SCANNER, LINEAR

FIGURE 3-4A

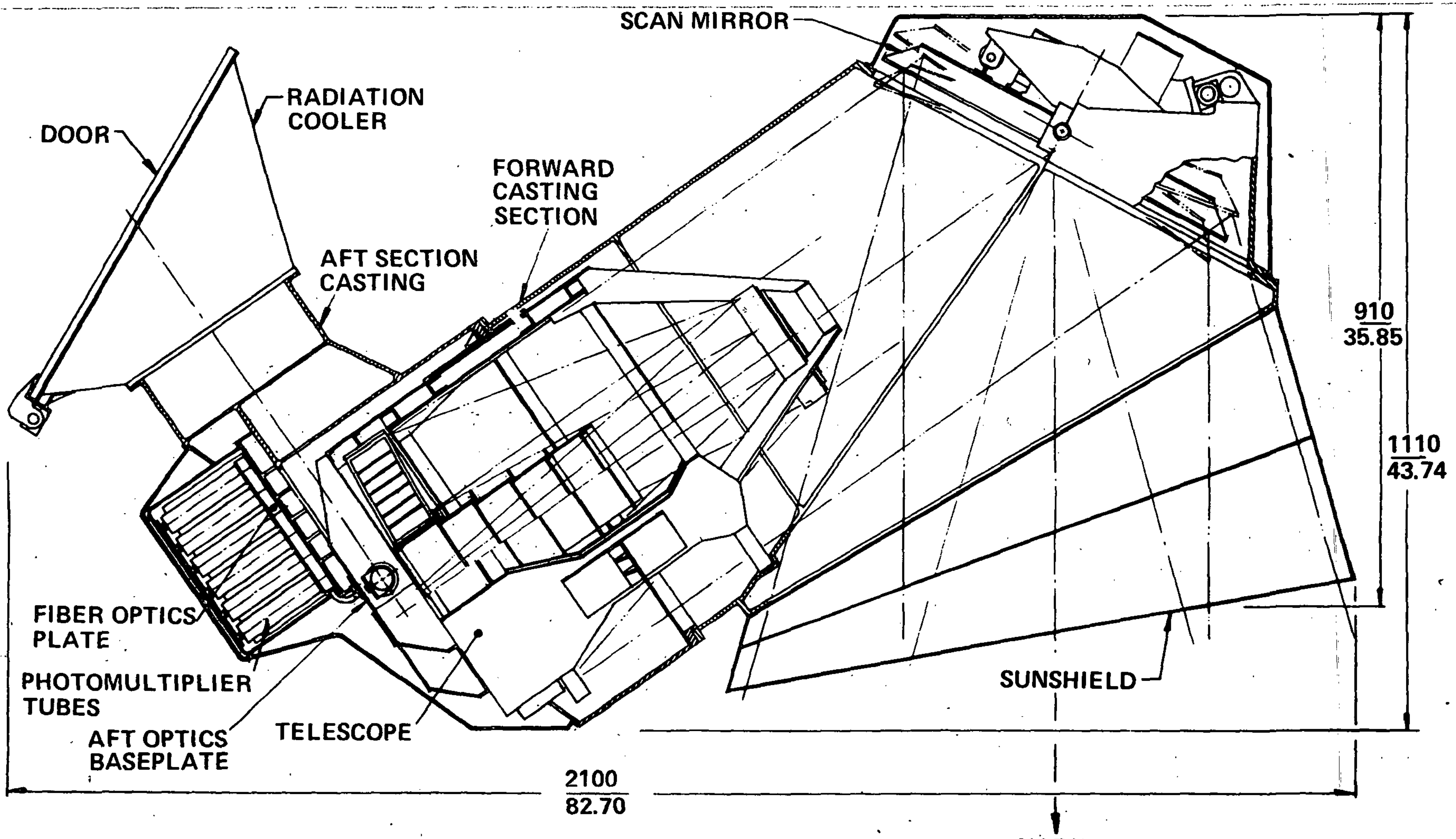


FIGURE 3-4B

THEMATIC MAPPER - OBJECT PLANE SCANNER

Parameters of the HRPI for the EOS-A/LRM mission are tabulated below:

Number of spectral bands	4
IFOV	10 meters @914-km altitude
Swath Width	48 km (25.9 n.mi.)
Volume	Not to exceed a cylinder 0.914 meters in diameter x 2.134 meters long (36 inch diameter x 84 inch long)
Weight	272.7 kg (600 pounds)
Power (exclusive of heaters)	100 watts
Real-time output bit rate	120 Mbps including overhead

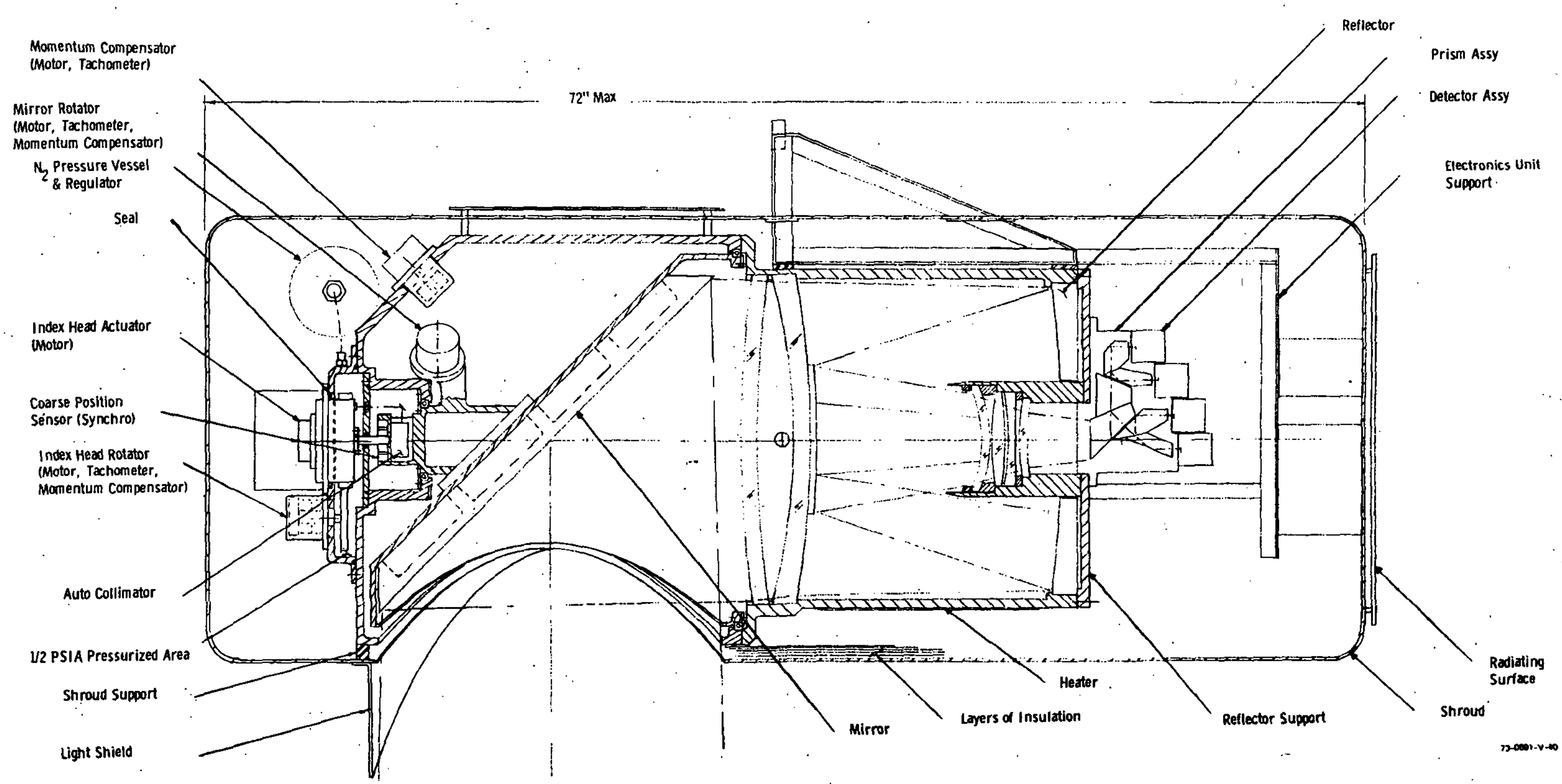
A point design of this instrument has been completed; a cross-section view is shown in Figure 3-5.

3.2.1.3 Data Collection System (DCS)

Various systems incorporating satellite collections of data from earth-sited sensor platforms have been investigated on programs other than EOS. Frequently the emphasis has been on a feature to provide the location of each platform as an adjunct to the data. The primary role envisioned for a Data Collection System on the EOS-A mission is monitoring of hydrologic data, for which platforms are located at known fixed sites. Emphasis will be on high traffic capacity of the satellite collection process, and on a design which will permit simplicity (hence low cost) of the platforms.

Parameters of the DCS for EOS-A are tabulated on the next page:

Frequency	VHF
Traffic Capacity	At least 200 platforms within a 370.4 km (200 n.m.) diameter cycle
Size	Not to exceed 0.3 x 0.3 x 0.6 meters (1 x 1 x 2 feet)
Weight	Not to exceed 35 kg (77 pounds)
Power	Not to exceed 50 watts



HIGH-RESOLUTION POINTABLE IMAGER, "PUSHBROOM" TYPE

FIGURE 3-5

3.2.1.4 Synthetic Aperture Radar (SAR)

The inclusion of active-microwave systems (i.e., radars) for remote sensing in an early EOS payload was specifically recommended by the EOSMRG. As is so often the case, the requirements for a radar addressed to geologic survey, land use monitoring, and water/ice monitoring differ markedly from the requirements imposed on a radar addressed to meteorological and oceanographic problems; so to exploit fully the potential of radar for R&D in the EOS era 1978-1981 requires two radars of significantly different characteristics. The M/O radar will be discussed elsewhere.

Parameters of an SAR currently being studied for follow-on land-resources management applications are tabulated below:

<u>Frequency</u>	Dual, X- and L-band
<u>Geometry</u>	
Altitude	914.2 km (493.6 n.mi.)
Depression Angle	70°
Swath Width on Ground	40 Km
<u>Resolution</u>	
Azimuth	30 meters
Range	18.8 meters (30 meters on ground)
<u>Antenna</u>	8.7 m x 0.914 m x 0.457 m
Dimensions	(28 feet x 3 feet x 1.5 feet)

3.2.1.5 System Performance Characteristics

System output quality requirements (as influenced by instrument characteristics, Table 3-A) are shown in Table 3-B.

TABLE 3-A

INTERIM DESIGN PARAMETERS FOR THE THEMATIC MAPPER (TM) AND HIGH RESOLUTION POINTABLE IMAGER (HRPI)

			Thematic Mapper		HRPI	
Band No.	Spectral Region (μm)	Assumed Radiance, N_1 $(wm^{-2}\ sr^{-1})$	IFOV (urad)	S/N(PP/RMS) (@N_1 &MTF=1)	IFOV (urad)	S/N(PP/RMS) (@N_1 &MTF=11)
*1	0.5 - 0.6	2.2	35	10	10	6
*2	0.6 - 0.7	1.9	35	7	10	6
*3	0.7 - 0.8	1.6	35	5	10	6
*4	0.8 - 1.1	3.0	35	5	10	6
5	1.55 - 1.75	0.8	35	5	-	-
6	2.1 - 2.35	0.3	35	5	-	-
7	10.4 - 12.6	20.0@ 300K	140	0.5K NEΔT @300K	-	-

*Spectral Bandwidth may be reduced.

TABLE 3-B

SYSTEM OUTPUT QUALITY REQUIREMENT

	Geometrically Uncorrected		Geometrically Corrected	
	TM	HRPI	TM	HRPI
Swath Width	185	48	185	48
Spatial Resolution (m)				
Visible	30	10	30	10
Thermal	120	--	120	--
Linearity (μrad)	0.2 IFOV	0.2 IFOV	0.2 IFOV	0.2 IFOV
Band to Band Registration	0.1 IFOV	0.3 IFOV	0.1 IFOV	0.3 IFOV
Position Accuracy (w/o GCP)**	±450m	±450m	±170m	±170m
Position Accuracy (with GCP)**	--	--	± 15m	± 15m
Radiometric Accuracy				
Visible				
Tape	± 1.6%	± 1.6%	± 1.6%	± 1.6%
Film	± 5%	± 5%	± 5%	± 5%
Thermal				
Tape	±1K	--	±1K	--
Film	±3K		±3K	

* Includes radiometric correction, earth-rotation correction, line-length adjustment, correction for earth curvature, and predicted emphemeris.

† Additionally includes use of best-fit ephemeris from measured data.

** GCP = ground control points.

3.2.1.6 INSTRUMENT MISSION PECULIARS

3.2.1.7 WIDE BAND DATA HANDLING

The high resolution sensors on EOS-A require a new data handling concept - The Multi-megabit Operation Multiplexer System (MOMS). As presently designed, the MOMS will contain a wide band analog multiplexer, high speed analog-to-digital converter and be capable of generating a composite bit stream of data derived from both the Thematic Mapper (TM) and the High Resolution Pointing Imager (HRPI). These two sensors will produce a data rate (including overhead) of approximately 240×10^6 bps. This data will be transmitted to user stations as a quadraphase signal at an equivalent rate of 120×10^6 bps. A secondary MOMS function will be to selectively produce data of reduced resolution or of reduced swath size and colors which will be available at rates of approximately 20×10^6 bps. Data of this type is expected to be most beneficial to local users who can operate with selectable forms of the lower data quantities but cannot afford the processing of 240×10^6 bps.

Synthetic Aperture Radar data handling is not completely defined, however transmission of the data in digital form is planned. Necessary analog-to-digital conversion and rate buffering hardware will be integral components of the SAR instrument.

3.2.1.8 WIDEBAND COMMUNICATIONS

Wideband sensor data will be transmitted to ground stations by use of the 8.025 to 8.4 GHZ frequency band. Normally two transmitters will be operated on the observatory simultaneously; one for transmitting thematic mapper and HRPI data to NASA stations and the second for transmitting selected data to local user terminals. Their designs will be identical and the output of either transmitter may be split between both antennas. Either transmitter may transmit any data. The transmitter design will use either a traveling wave tube or solid state technology with the choice being based on cost, efficiency, and reliability.

Two high gain x-band antennas, steerable through a field of view of 120°, will be located at the forward end of the observatory.

Communications system link calculations indicate a gain requirement for both antennas of 30 db (5° beam). At 8 GHZ, this gain can be produced by parabolic reflectors with apertures of 1.6 feet. Antenna pointing requirements can be easily met by periodic X-Y updates from the on-board computer. Granularity and frequency of updates will be sufficient to assure smooth antenna motion.

3.2.1.9 INSTRUMENT STRUCTURE/THERMAL

The configuration for the instrument structure will vary from mission to mission as a function of the instrument complement.

As an example, for a Titan or Shuttle EOS configuration depicted in Figure 3-6A), individual instruments are supported or housed within an instrument support structure. This structure is supported directly from the top surface of the transition ring. It consists of an open tubular truss system for the purpose of minimizing gradients through internal cross fire radiation. Attached to the outer surface of the rectangular structure is a primary thermal shield composed of an Alzak skin backed by a multi-layer thermal insulation blanket. Internal to each instrument cavity, heater power will be used to maintain a constant overall internal temperature. Each instrument placed in the vacity will be insulated on the three internal facing surfaces (with the exception of low temperature cooler areas). The fourth earth facing surface of the instrument will be the primary heat dissipating surface for the instrument.

For a Delta configuration (Figure 3-6B) with only one or two instruments flying on one mission, a primary shield to damp out the effects of transient fluxes on instrument supporting structure may not be needed since the instruments may be attached directly to the transition ring (no resupply) or to short canti-levered structures which can be individually insulated. However, even in this case the instruments must be insulated to minimize distortion effects of time varying environmental fluxes.

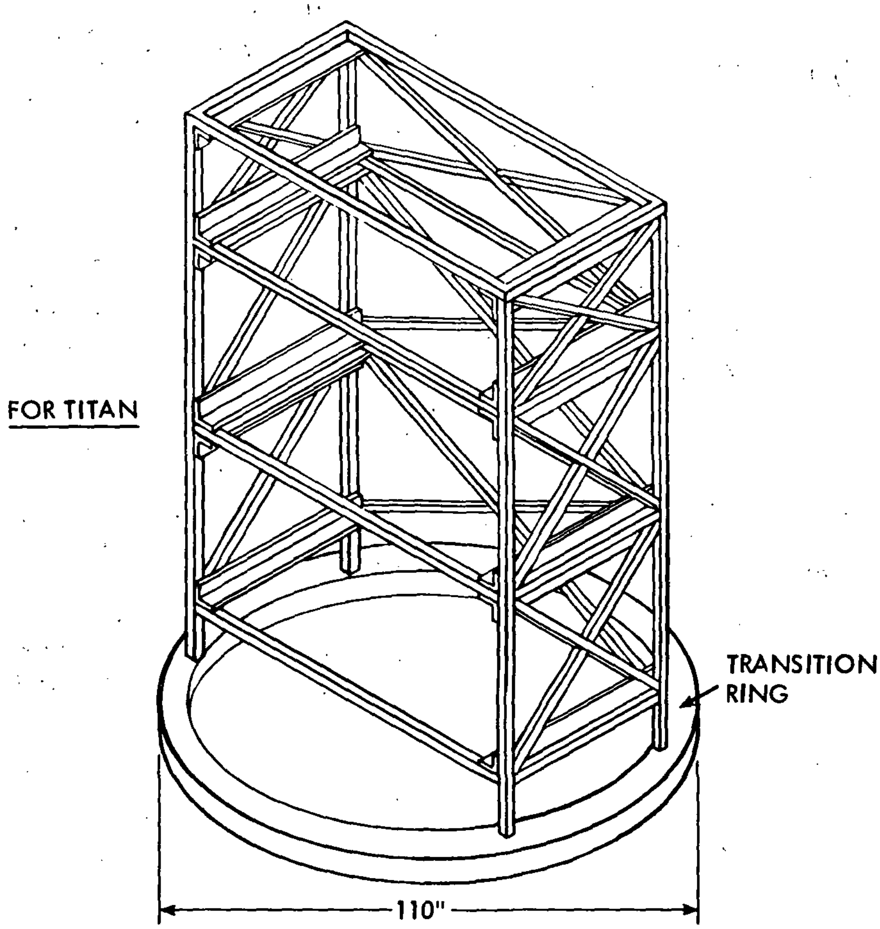


FIGURE 3-6A

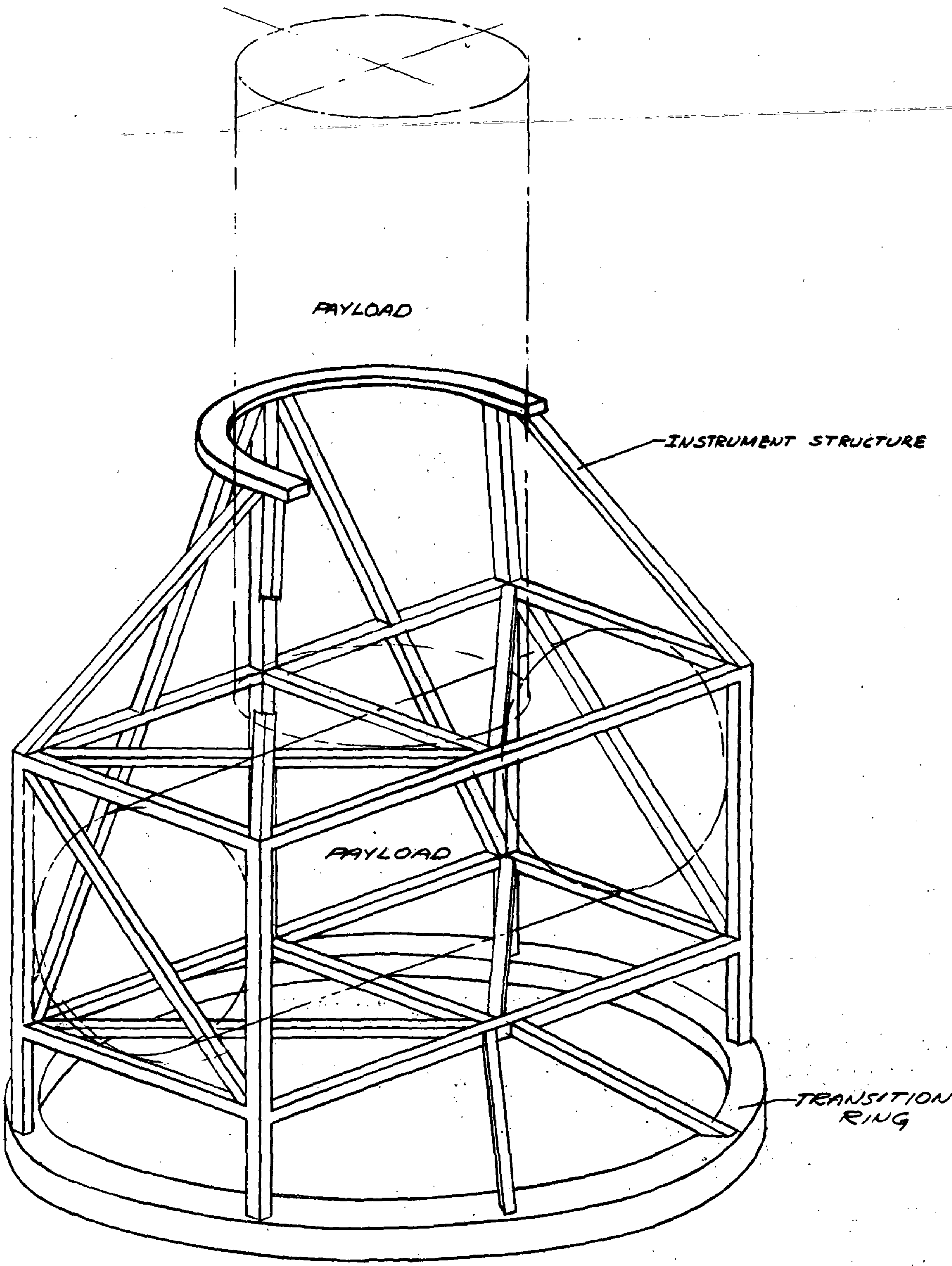


FIGURE 3-6B

3.2.2 SPACECRAFT

3.2.2.1 SPACECRAFT SUBSYSTEMS

The spacecraft has been divided into discrete subsystems modules in order to provide:

a. A spacecraft design that is insensitive to the payload size.

b. Compatibility with the Delta, Titan and shuttle launch vehicles.

c. Capability for eventual in-orbit maintenance at the subsystem level.

d. Capability for eliminating multiple environmental tests at the component level in favor of subsystem level testing.

e. A design that is sufficiently insensitive to its placement in the system that the subsystem level tests are realistic. This will allow eventual elimination of environmental testing at the system level.

f. The option for commonality without restricting legitimate mission requirements by allowing systems integration to be a responsibility of the unique missions.

The point design subsystems described here are only one way to meet the performance requirements. Subsystem procurement will be on the basis of performance, interface and modularity specifications. Contractor will be encouraged to discover lower cost techniques than used in the internal point designs and management approaches for existing components.

3.2.2.1.1 COMMUNICATIONS AND DATA HANDLING

The C&DH subsystem will provide a means for observatory tracking, ground and on-board control of all spacecraft and instrument functions, and for retrieval of observatory status data. An S-band communications subsystem, consisting of a transponder and an omni-directional antenna, will be used for ranging, receiving commands, and transmitting the narrow band telemetry format which will contain housekeeping, DCS, and PMI data. The command decoding and telemetry data acquisition hardware will be interfaced with an on-board computer to form an integrated data handling function. The computer will communicate with all observatory subsystems and perform functions such as attitude control, thermal control, stored commands, and antenna pointing.

The data handling design is based on the concept of remote multiplexing of telemetry data and remote distribution of commands with the remote hardware located in the user subsystem and instrument modules. Command and telemetry data will be routed to and from the C&DH module in serial digital form to minimize the number of interconnections in the resupply connectors.

Determination of spacecraft position will be achieved through the use of both ground truths and range and range rate (R&RR) data. The S-band transponder will use coherent side tone frequencies for ranging and STDN engineers forecast that resulting range measurements will be sufficiently accurate to produce knowledge of position along track and across track to within 30 to 50 meters. Studies show that by combining this information with ground truths, determination of position to the desired accuracy is achievable.

Spare space in the C&DH module will be used for housing the Data Collection System and the tape recorder for the Pollution Measurement Instruments. A point design configuration and functional/interface block diagram are shown in Figure 3-7a and 3-7b.

3.2.2.1.2 POWER

The power subsystem will be housed in a standard subsystem module approximately four feet square by eighteen inches deep with radiation from a 4'x4' surface. It will be capable of storing, distributing and monitoring power derived from a solar array. The physical and electrical characteristics of the arrays will be based on specific mission requirements.

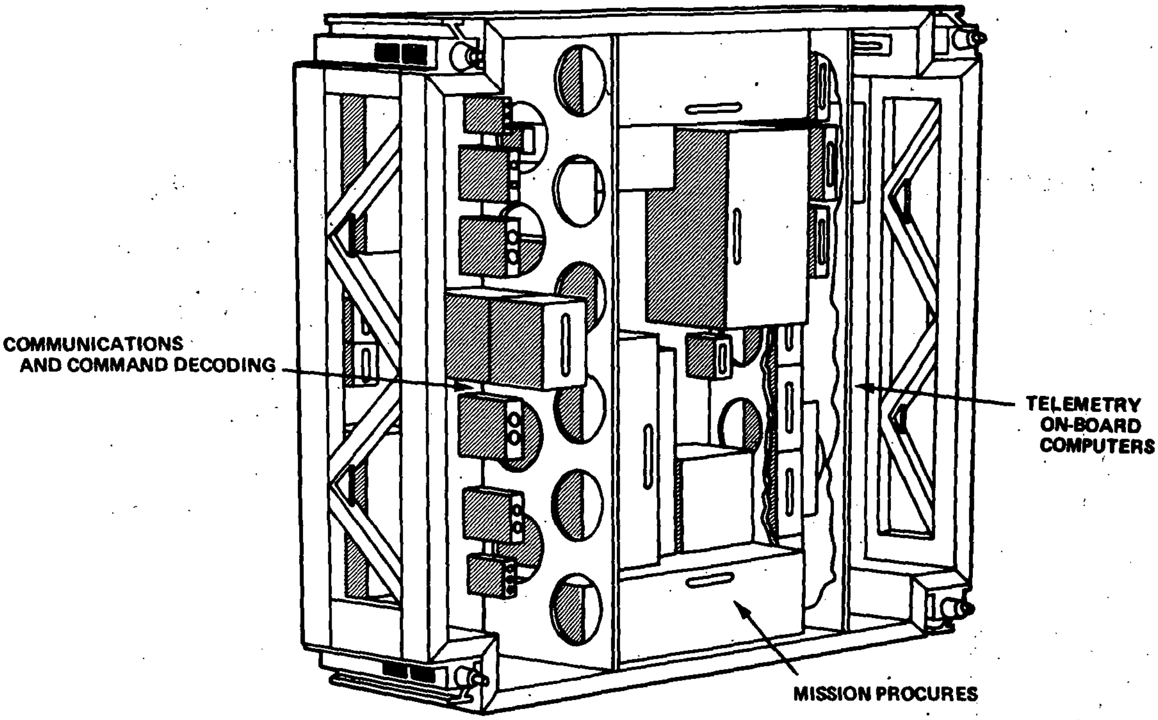
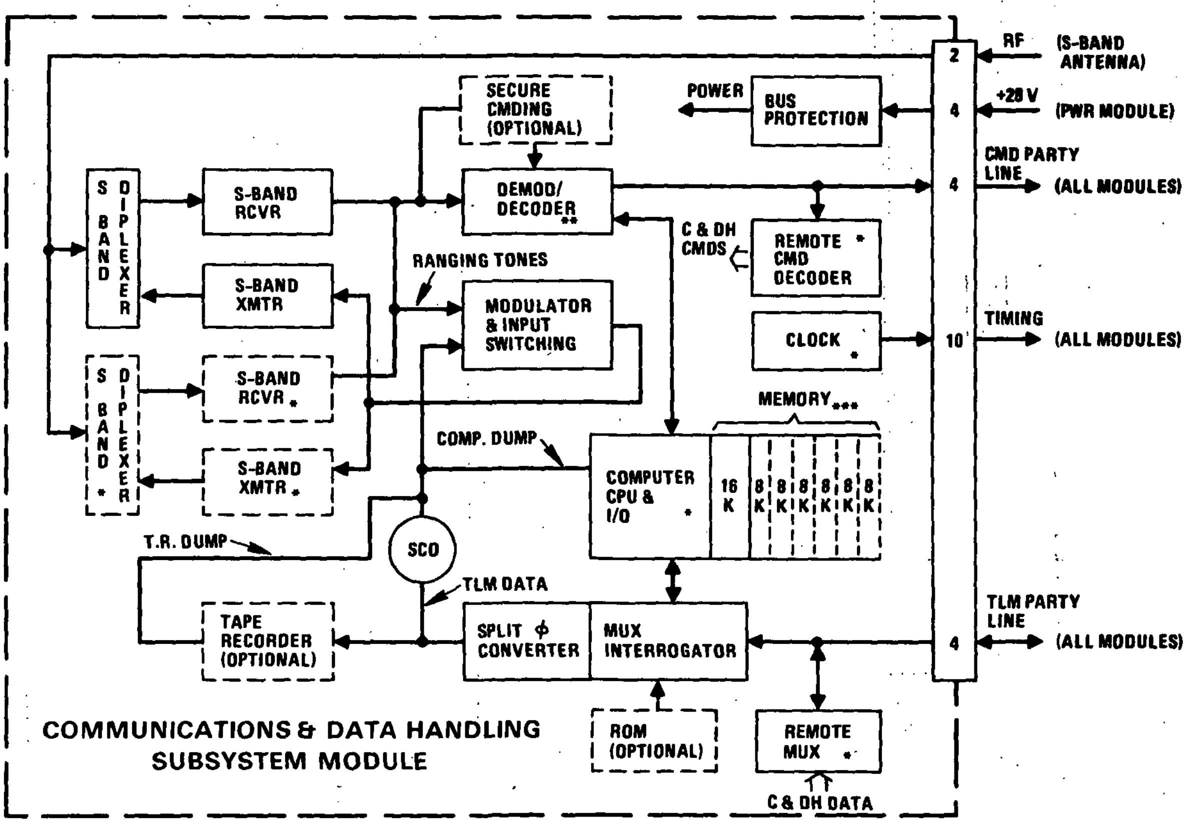


FIGURE 3-7A



* OPTIONAL REDUNDANCY
** OPTIONAL REDUNDANCY/DIVERSITY COMBINING
*** OBC MEMORY SIZE FROM 16K TO 64K WORDS IS MISSION UNIQUE

FIGURE 3-7B

C&DH MODULE

The module will contain one or more nickel-cadmium storage batteries, each with a capacity of 20 ampere-hours. The maximum battery capacity of the module will be at least 120 ampere-hours but the number of batteries used on a specific mission would be based on mission requirements such as type or orbit, peak and average power requirements, life, etc.

Command and telemetry interfaces to the communications and data handling subsystem will be via remote decoders and multiplexers which will provide all operational interfaces with ground control and the on-board computer.

Power output to the payload and other subsystems will be plus 28 volts. Bus regulation will be plus or minus seven volts or better. A point design configuration and functional/interface block diagram are shown in Figure 3-8a and 3-8b.

3.2.2.1.3 ATTITUDE CONTROL MODULE

The ACS will provide for control of the spacecraft during initial acquisition, reacquisition and normal operations. Initial acquisition and re-acquisition modes keep the solar array aligned normal to the sunline. For the normal operating mode, the ACS will be capable of pointing the observatory continuously to the earth's centroid to an accuracy of better than .01 degrees. Attitude perturbation will be less than 2 arc seconds and the rate of attitude drift will be less than $2x10^{-6}$ degrees per second.

The approach in the ACS design has been to adapt proven techniques and, where possible, flight qualified equipments to a modular design concept. The basic control system element is a high performance gyro attitude reference system with the attitude reference periodically updated with a fixed star tracker. A general purpose computer, located in the C&DH module, will perform sensor data processing and control functions such as mode switching logic, momentum wheels, magnetic unloading and jet firings. A point design configuration and functional/interface block diagram are shown in Figure 3-9a and 3-9b.

A pneumatics module will be housed at the rear of the spacecraft and will be used to provide the torque required for initial stabilization maneuvers. This consists of orienting the observatory in the safety (or sunbathing) mode. The pneumatics system will also be used as a backup momentum unloading system.

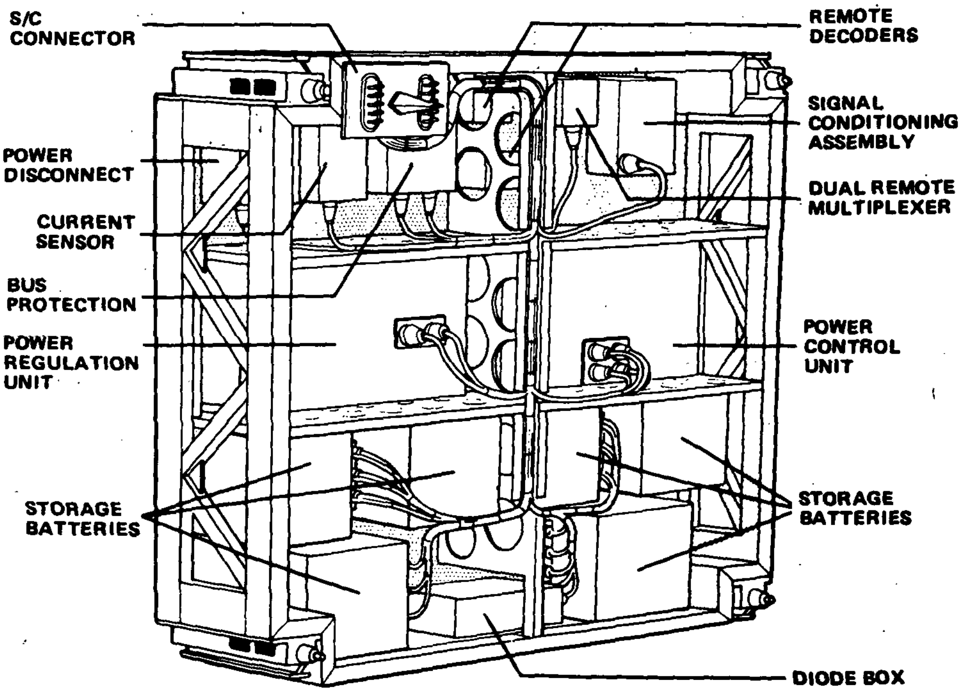


FIGURE 3-8A

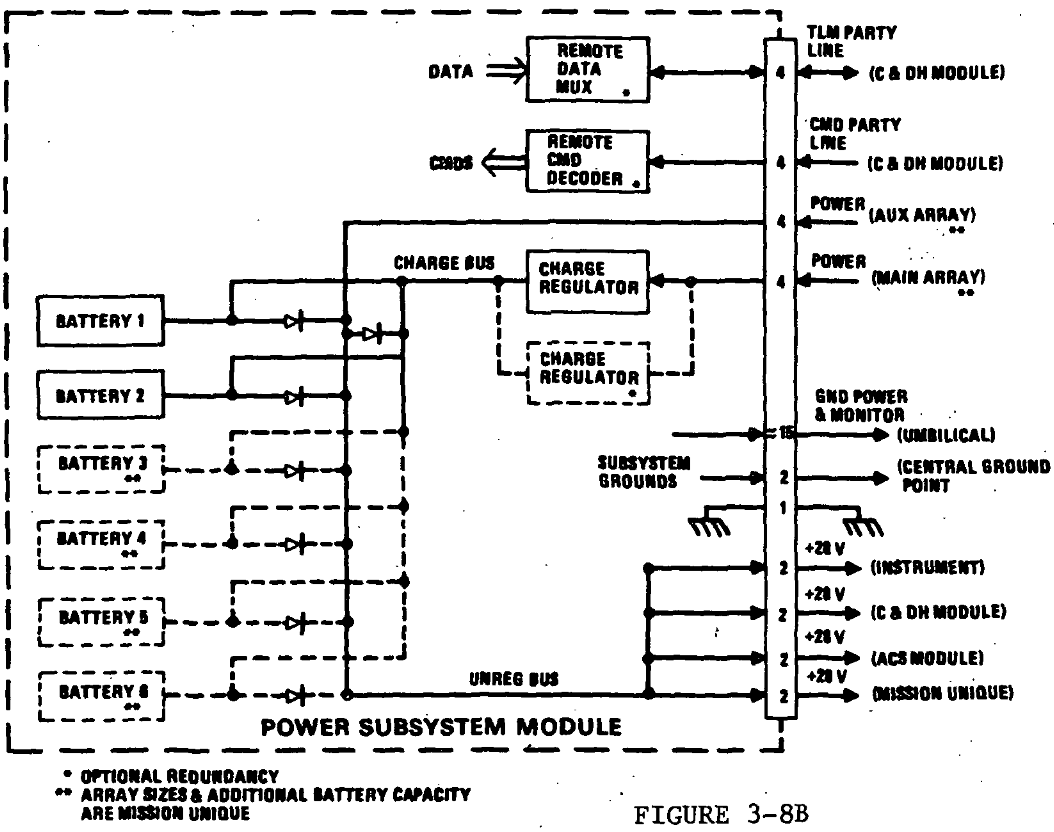


FIGURE 3-8B

POWER SUBSYSTEM MODULE.

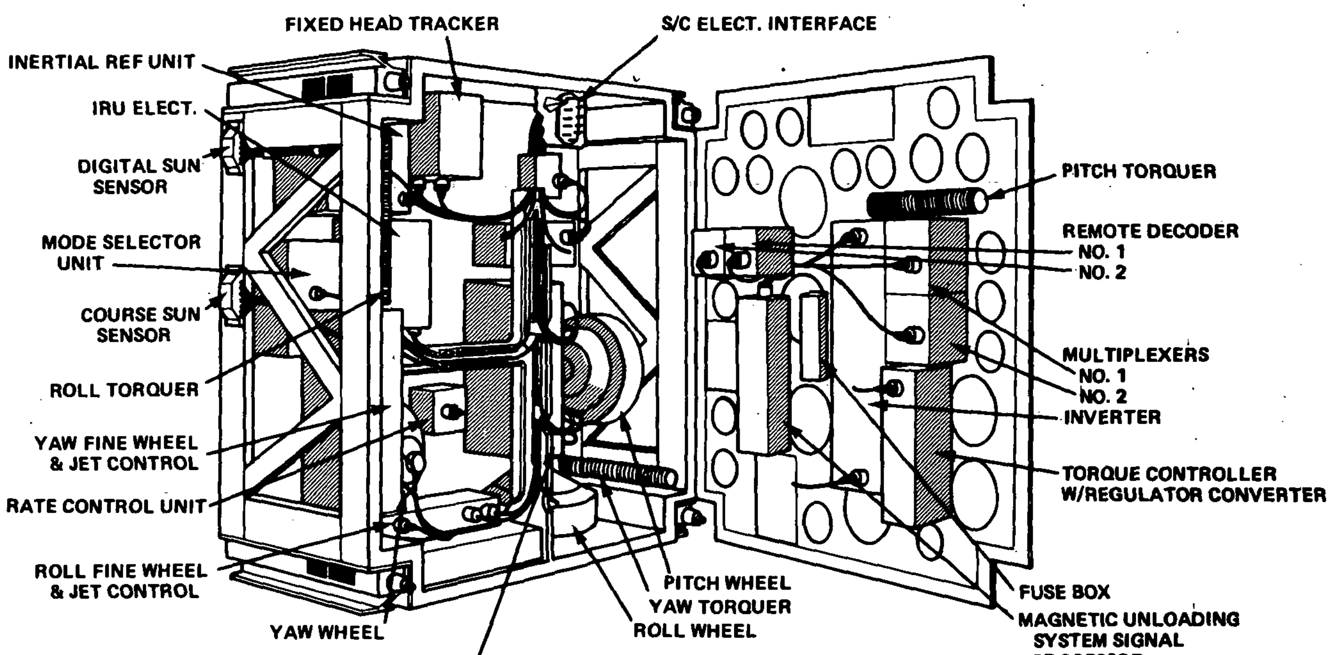
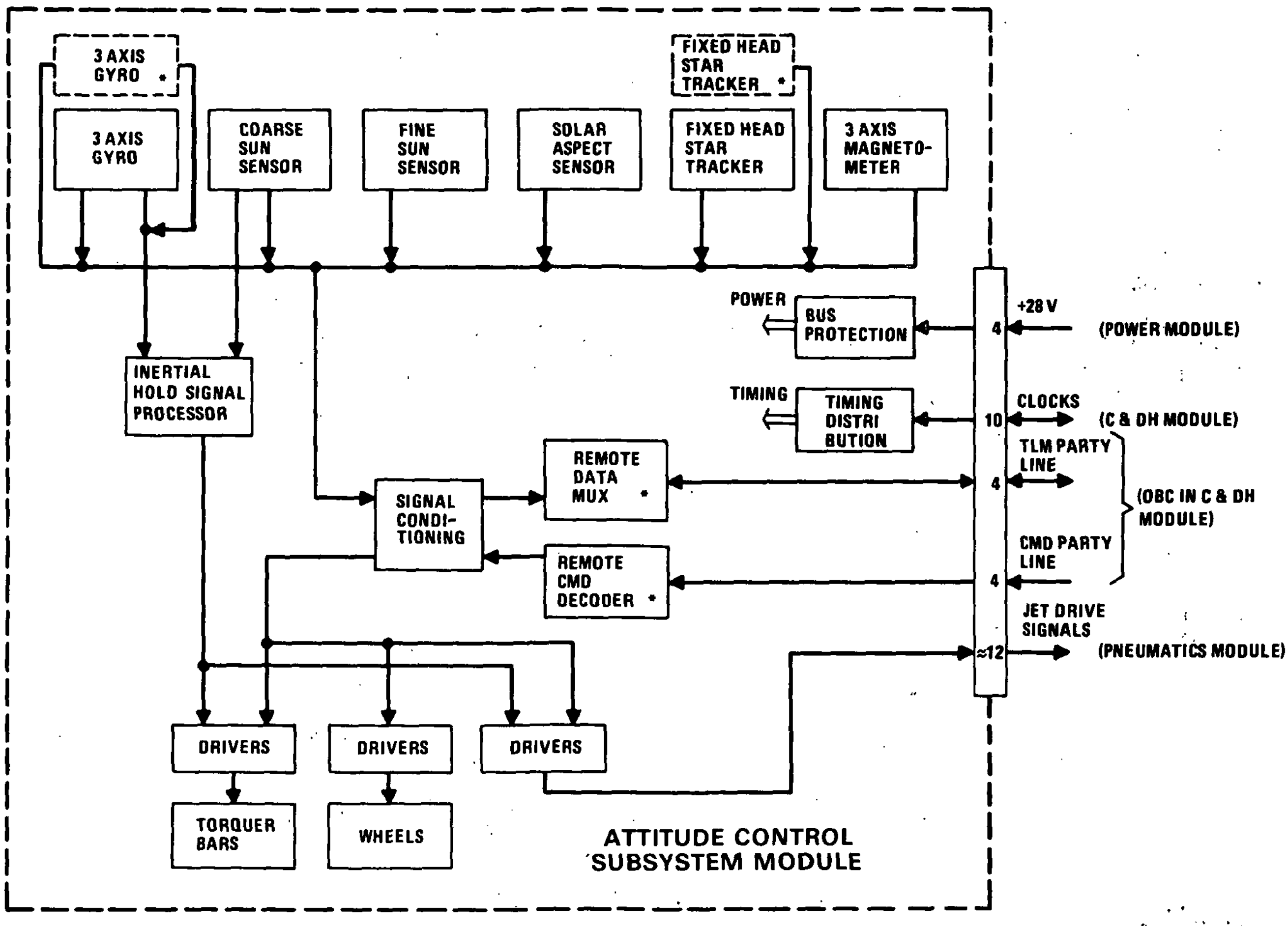


FIGURE 3-9A



*OPTIONAL REDUNDANCY

FIGURE 3-9B

ATTITUDE CONTROL SUBSYSTEM

3.2.2.1.4 MECHANICAL SUBSYSTEM

The three subsystem modules are physically the same size measuring 18 inches deep, 48 inches wide and 48 inches high, and have the same basic external structure as shown in Figures 3-7A, 3-8A and 3-9A

Each corner of the module frame is relieved to a depth and width of 4 inches to accept a resupply latching/release mechanism. A structural heat sink panel is fastened to the main frame and closes out the outboard looking 48" x 48" surface of module structure. The majority of subsystem components, expecially those which are heavy and/or are high heat dissipators, are mounted directly to this outboard facing panel. The launch reaction loads of subsystem components are transferred from the panel through the main structural frame through the assembly structure to the transition ring. Each subsystem module is designed to carry the full launch reaction loads of all its subsystem components.

Although the frame and resupply mechanisms design is identical for all three electronic modules, each module can be structurally tailored internally to facilitate the packaging peculiarities of each subsystem. This is accomplished through selective placement of the vertical and horizontal bulkheads within the frame and by the selective design modification of the outer panel to accommodate star tracker "see through" requirements and any local thermal tailoring or special dissipating heat sinks.

3.2.2.1.5 THERMAL

Thermal control will preferably be achieved with passive designs utilizing thermostatically controlled heaters to maintain temperatures in the range 70°±20°F (except for batteries at 40°F and the IRU at 140°F). Modules will be insulated on five faces with multilayer insulation; the remaining outboard face will serve as a heat sink which will be radiatively coupled to a flight thermal control skin for rejection of internally dissipated component power as shown in Figure 3-5. Structures and instruments will also be insulated and controlled at nearly constant temperatures to gain thermal independence of all spacecraft elements. Similarly, commonality of thermal design will be attained for a wide range or orbits, altitudes and pointing requirements. Only simply made heat sink emittance changes will be necessary for changes of missions.

3.2.2.1.6 Basic Software

There will be furnished with the on-board computer a set of basic software which will be applicable to each mission. This software will be modular so that mission unique functions such as instrument control, booster control, and antenna pointing may be easily added to the flight program. The basic programs are an executive which handles I/O and task management; a stored command handler; and the structure for a status buffer handler which reports observatory back-orbit status and thus obviates the need for a status data tape recorder.

The program listed above will be developed for the Advanced On-Board Computer (AOP) in the demonstration spacecraft and no conversion would be required for their use in flight for those missions which use AOP's. Various other subsystem programs such as thermal control, power regulator control, momentum unloading, gyro update, etc., will be developed for the demonstration spacecraft and thus will be available for use in flight. These programs, however, are dependent on the final design of the power and ACS modules and may require modification.

There exists a software development facility for the AOP with a breadboard AOP connected to a general purpose ground computer. The support software package is in Fortran so that it approaches being machine independent.

With the software development facility and the flight exec being designed to easily accommodate the deletion, addition, and checkout of application programs, both subsystem and mission unique programs can be simply and reliably added to the flight program.

Thermal control of the subsystem support structure is achieved by insulating all space exposed surfaces with low α/e skins backed with multilayer insulation. The subsystem modules, whose normal operating set point is $70^{\circ}F \neq 20^{\circ}$, completely surround the truss sections of the structure. A limited amount of heat is conducted from the modules through the resupply mechanisms to the structure. The structure will therefore be near the average operating set point of all the surrounding module. In normal operations the absolute temperature level of the structure is not critical, however, it must stay constant and the associated temperature gradients must remain constant. For this reason all internal structural members are insulated to assure long time constants.

During shuttle resupply operations, in order to avoid excessive friction in the resupply mechanism, the subsystem support structure would be near the mean temperature of the subsystems. This will be achieved by the use of make-up heaters attached to the structural members. Power for these heaters will be provided from the shuttle.

3.2.2.2 Spacecraft Mission Peculiars

3.2.2.2.1 Subsystem Support Structure

For the Titan/Shuttle configuration the subsystem assembly structure contains four resuppliable subsystem modules and the PMI module. Four of the modules are contained between the upper and lower decks. They are the power subsystem module, the stabilization and control module, the stabilization and control module, the communications and data handling module and the Pollution Monitoring Instrument Module. The fifth module is the pneumatics/orbit adjust module and it is attached to the underside of the assembly structure. The basic function of the subsystem assembly structure is as follows:

a. To support the lateral and thrust loads of the modules during launch and docking.

b. To maintain alignment between stabilization and control subsystem module and the reference axis of instrument payload assembly during orbit operations.

c. To act as a holding device for resupply release/restraint fittings, the system electrical harness, the subsystem electrical connector, the power distribution box and the thermal insulation panels.

The subsystem support strucuture is a welded aluminum assembly as shown in Figure 3-10A which derives some of its load carrying capacity through the modules which it supports. The support structure transfers these module loads directly to the transition ring attached to the upper deck. For launch vehicle thrust loads the structure reacts in tension. For lateral loads the structure which is cantilevered from the transition ring receives its bending stiffness from the modules.

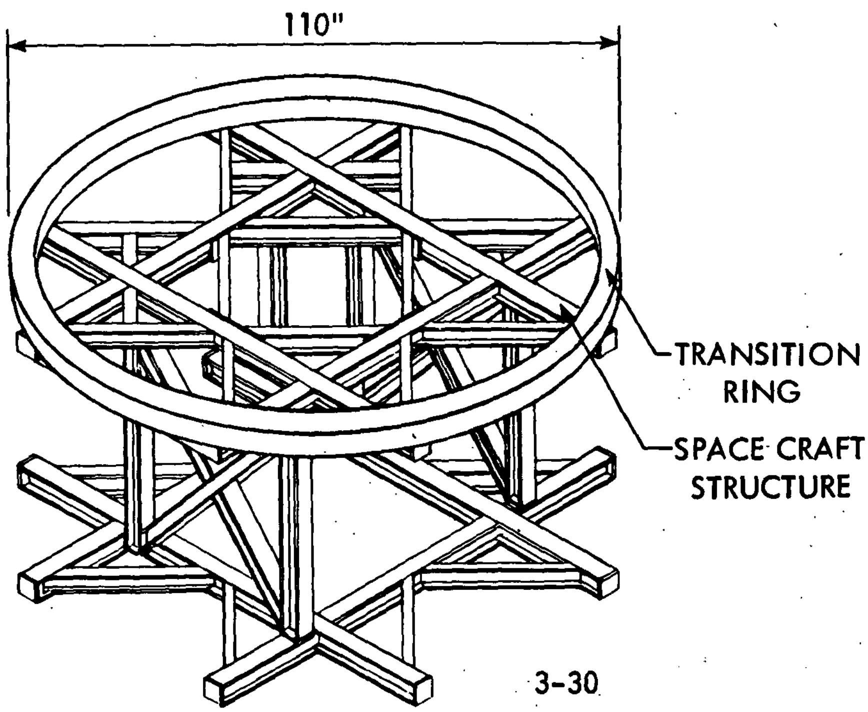


FIGURE 3-10A

EOS/TITAN CONFIGURATION.

For the case where the EOS mission is constrained to be launched by the Delta launch vehicle, the subsystem packaging configuration is significantly different although the identical subsystems are used. In this situation the three subsystem modules are placed in a triangular configuration, as shown in Figure 3-10B.

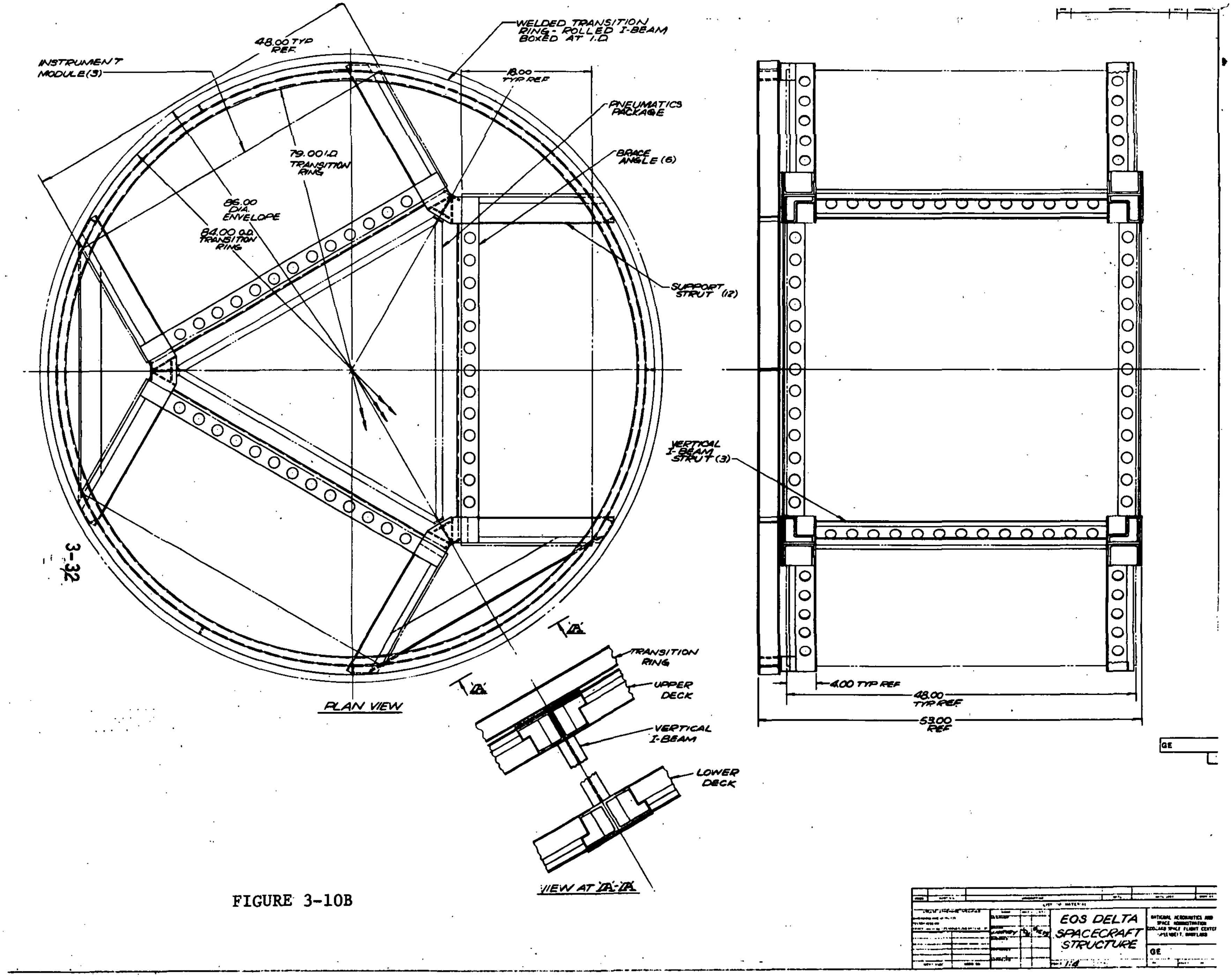

INSTRUMENT MODULE (3)
48.00 TYP REF.
WELDED TRANSITION RING - ROLLED I-BEAM BOXED AT I.D
18.00 TYP REF
PNEUMATICS PACKAGE
BRACE ANGLE (6)
79.00 I.D TRANSITION RING
86.00 DIA. ENVELOPE
84.00 O.D. TRANSITION RING
SUPPORT STRUT (12)
VERTICAL I-BEAM STRUT (3)
PLAN VIEW
TRANSITION RING
UPPER DECK
VERTICAL I-BEAM
LOWER DECK
VIEW AT A-A
4.00 TYP REF
48.00 TYP REF
53.00 REF
GE
EOS DELTA SPACECRAFT STRUCTURE
1:4

FIGURE 3-10B

The ACS module is located on the anti-earth side and the power subsystem is 120^{0} from it on the anti-sun side. The C&DH subsystem closes out the triangle. The pneumatics/orbit adjust package occupies the central triangular space formed by the three subsystems.

The primary load carrying members are the structural frames of the subsystem modules. Each external subsystem frame is attached to the adjacent modules. Upper and lower corner braces are attached to the 4"X4" relieved corners of the modules (in place of the resupply mechanisms) to provide torsional and lateral stiffness. The pneumatics/orbit adjust module is also tied directly to the transition ring through which launch reaction loads are transferred.

The Delta configuration spacecraft is capable of carrying a full Delta shroud diameter payload, in which case an 86" diameter transition ring is used.

It is also capable of carrying a weight constrained five foot diameter payload. For this latter case, the EOS is launched upside down with the payload attached to a 60-inch diameter transition ring and sitting inside the launch vehicle interstage adapter.

3.2.2.2.2 Pneumatics-Orbit Adjust Module

The pneumatics/orbit adjust module consists of two cylindrical sections which are bolted together as shown in Figure 3-11. The first section contains the pneumatics subsystem with up to eight spherical N_2 tanks, all associated piping, manifolding, regulator and solenoid actuated values. It is thermally isolated from space with multi-layer insulation blankets behind outer Alzak skins. At the outer lower periphery of this section is a system of 16 gas jets to effect changes in spacecraft momentum in pitch, yaw and roll axes. The jets protrude radially just beyond the outer thermal skin. All other components including the gas solenoids are internal to this section.

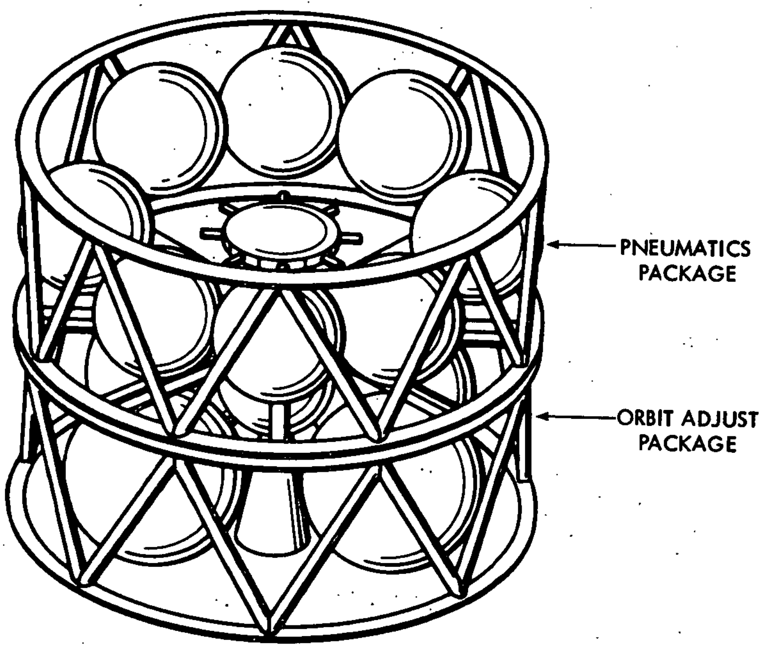


FIGURE 3-11

Since not all missions require an orbit adjust package some missions will fly with only the pneuamtics section. On the other hand, those missions requiring orbit adjust can bolt on this mission peculiar package to the bottom of the pneumatics package. Either the pneumatics package by itself or the combined package can be shuttle resupplied.

For those missions requiring an integral tug (as discussed in Section 3.2.4), it is possible to house the tug engines within the orbit adjust/ pneumatics module. In this configuration the thrust vector control and orbit adjust functions can be combined and the pneumatics system including the N_2 tanks are housed within this same module as shown in Figure 3-12. Again in this configuration the module thermal design concept is identical to that of the pneumatics and orbit adjust sections discussed above.

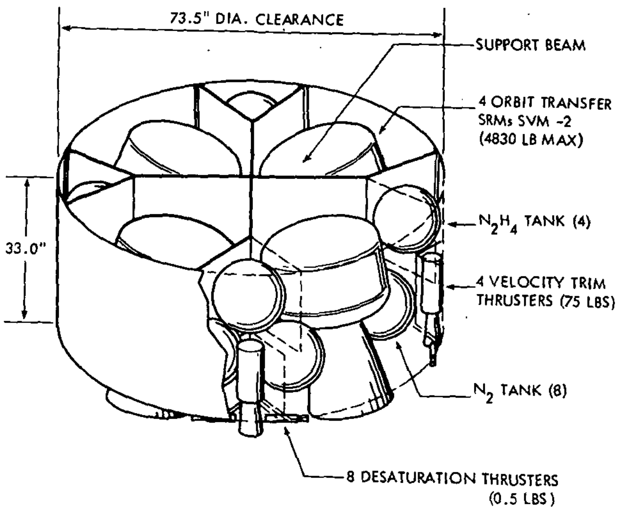


FIGURE 3-12

3.2.2.2.3 Reaction Wheel/Magnetic Torquer Options

The Attitude Control Subsystem has the capability of accommodating a series of reaction wheels that cover a wide range of momentum storage (2 to 100 foot-pound-seconds) and torque levels (2 to 50 inch-ounces). This capability allows this subsystem to meet the performance requirements of a wide range of spacecraft sizes, weights and configurations and, in addition, accommodate various disturbance sources in the payload equipments. The various reaction wheels shall be completely interchangeable electrically and with regard to mounting arrangement (except for envelop clearance requirements).

The reaction wheels are sized with regard to the cyclic disturbance torques. In a like manner, a series of interchangeable torquer bars are provided to compensate for the secular components of the disturbance torques.

3.2.2.2.4 Battery Capacity Options

The power subsystem module will be designed for adaptability to special mission requirements. The subsystem will be designed to accommodate a variable number of batteries according to mission load and life requirements. Battery capacity will be selectable in steps of about 20 ampere-hours up to a maximum capacity of 120 AH as governed by the volumetric and thermal dump capabilities of the module. The associated charge control and monitoring circuitry for the batteries will be nonredundant with space and harnessing for a redundent unit as dictated by mission requirements.

3.2.2.2.5 Computer Memory Options

In keeping with the concept of both basic and mission unique on-board computer software, the computer will have a basic memory capacity that can be augmented to meet the demands of each mission. A memory module will have a capacity of either 4096 or 8192 words and thus the computer memory will be expandable in those increments. The memory system design employs power switching so that the addition of memory modules is a cost, size, and weight but not a power consideration. The exact definition of which computer functions will be common to all missions has not been made. It is estimated that the basic memory size will be in the range of 16K to 24K words depending mainly on whether or not the computer usurps the function of a status data tape recorder for all missions. With a memory capacity of 24,576 words to handle mission independent functions in the computer, up to 40,960 words of memory may be added in 4K or 8K modules as the maximum capacity will be 65,536 words. It is felt that the large memory capacity will provide for efficient observatory operation without the attendant "shoe horn" problems normally associated with memory limited programs.

3.2.2.2.6 Mission Software

The on-board computer with its telemetry and command interface will be able to perform mission unique functions in both the spacecraft and the instrument areas. As mentioned, in Section 3.2.2.1.6, there will be a basic software system that is designed to allow the efficient and reliable addition of mission unique software. Instrument sequencing is normally handled by delayed commands in an open loop fashion. The computer can, of course, duplicate that activity. It is more likely, however, that instrument control will be accomplished by the computer in a more autonomous manner to reduce ground operations costs. The details of which instrument control functions are delegated to the computer will be decided by the system integration contractor. Similarly in the spacecraft area, the control laws for the ACS function, the philosophy and implementation of computer initiated emergency corrective action, etc. are systems questions to be decided by the systems contractor. The important point is that the computer as implemented on the observatory is a powerful tool and mission unique software may be added as modules to point a high-gain antenna, to provide thermostatic functions, to control battery charge rates, and to accommodate a host of observatory functions.

3.2.2.2.7 Solar Arrays

The solar arrays will be designed to meet specific mission requirements. The design will be based on the use of modularized sections which may be grouped to provide the required power output. The actual configuration of the array may range from one large assembly to two or more assemblies and may be either fixed or driven. Selection of fixed or driven arrays will be based on type of orbit, reliability requirements, cost effectiveness, etc.

The selection of solar cell materials will be based on required array life, type of array required (foldable or roll up) and other specific mission requirements.

A typical array for a two year or shorter mission would consist of 2X2 or 2X4 centimeter silicon cells about 12 mils thick, with a base resistivity of 2 ohm-centimeters.

For longer missions the use of 10 ohm-centimeter cells may be desirable. The cells would be protected with a 6 mil coverslide of Corning 7940 fused silica bonded to the cell surface. The cover slide external surface would have an anti-reflection coating to increase transmission characteristics and the internal surface would have a blue filter to protect the adhesive from ultra-violet exposure. The power output would be approximately 10 watts per square foot at 30 degrees centigrade.

3.2.2.2.8 Transition Ring/Interstage Adapter

Since payload requirements vary for each mission, the EOS subsystem configuration will be designed to be compatible with either the Delta or Titan IIID launch vehicles and the space shuttle. The key to this payload growth and flexible configuration capability is the unique concept of a transition ring. The instrument assembly and subsystems are structurally and thermally independent of each other with only one common bond, the transition ring. It is the transition ring that provides flexibility and adaptability in the selection of different types of launch vehicles. By modifying or replacing the type of transition ring, for each specific mission, the spacecraft subsystems can be used without redesign.

By the same token, since instrument assembly loads are transferred directly to the launch vehicle through the transition ring, any configuration change and/or weight growth of the instrument assembly can be accommodate by a simple change of the transition rings.

Designed to carry the weight of subsystems and instrument modules in parallel under various launch conditions, this transition ring is the structural interface which permits Titan and Delta launched missions to be retrieved by the shuttle. During the pre-shuttle launches of the EOS missions, the transition ring is tied to the launch vehicle through the interstage adapter. For a given mission the interstage adapter passes the reaction loads of the instruments and subsystems (in parallel) directly from the transition ring to the top of the launch vehicle as shown in Figure 3-13.

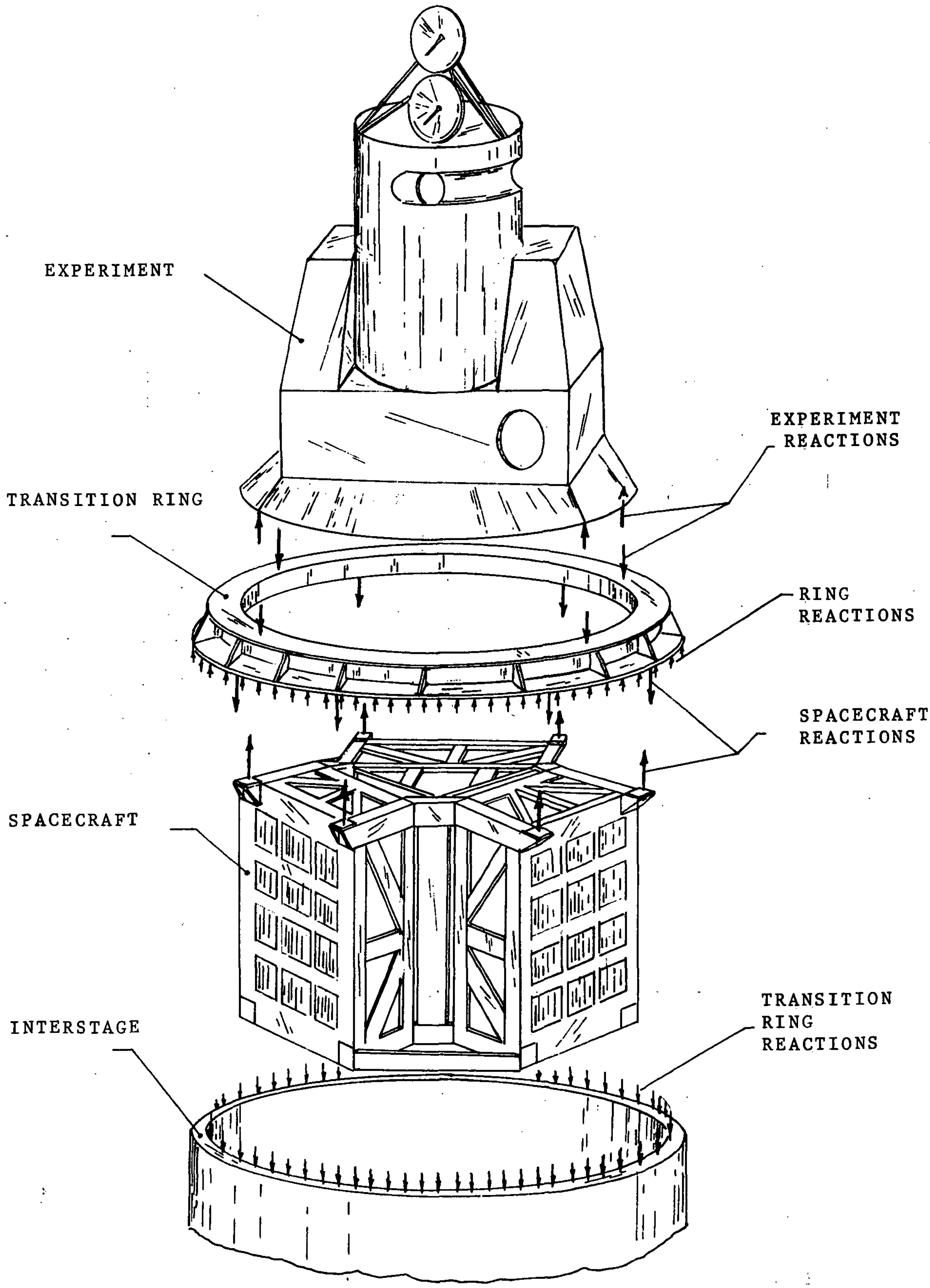


FIGURE 3-13

EOS TRANSITION RING SUPPORT CONCEPT

For each mission it is tailored to suit both the mission and the particular launch vehicle selected.

In the case of the shuttle, both in launch and retrieval situations, the transition ring is locked into the retention ring assembly. The reaction loads of subsystems and instruments are again cantilevered forward and aft of the transition ring as in the conventional launch vehicle case.

3.2.3 Data Management

3.2.3.1 System Considerations

Effective utilization of remotely sensed earth resource data depends upon an efficient data management system. Such a system must consider;

1. The effect of time delays in data availability.

2. Decentralized data processing.

3. Direct data transmission to local users.

The first consideration recognizes that information is perishable by nature and its value to a user is usually time dependent. Delays occur as a result of transmission, processing, reprocessing when necessary and dependence on external information (e.g orbital information) to aid in the data processing. Improved spacecraft stability will minimize processing delays and direct transmission of data to users will eliminate other delays.

Decentralized data processing is essential to the growth potential of EOS and any subsequent operational Earth Resource systems. National and international interest and involvement is increasing. Adding to this the increased sophistication of sensors and the accompaning increased data rates dictates decentralized data processing. A single central facility will be limited in quantity and different types of processing, particularly special processing. Decentralized processing shares the total work load and allows for discipline related facilities to evolve resulting in improved efficiency.

Direct data transmission to users has an impact on the system design of the space borne portion of the system. This capability, however, allows the local user, with a modest investment in a ground system, to receive data of local interest directly from the spacecraft. The advantage is one of immediate data availability and the use of a facility responsive to the user's particular needs. Dependence upon a central data collection and processing facility is only for more sophisticated processing.

3.2.3.2 Operational Control Center

The OCC is the focal point for EOS mission operations. It is through the OCC that commands are issued to the spacecraft in support of data investigation requirements and spacecraft system requirements. Trend analysis and mission planning are key functions within the control center and adjacent work areas. The OCC interfaces with the STDN and provides the data processing area with a real time source of data from the NTTF.

Prime functions of the OCC are:

1. Provide day-to-day planning and scheduling of the mission.

2. Direct central and remote sites in execution of the mission.

3. Contact the spacecraft via direct and remote high speed links.

4. Evaluate performance of instruments and spacecraft and adjust the operation accordingly.

5. Control and manipulate the instruments and spacecraft systems to execute the mission requirements.

3.2.3.3 NASA Data Processing Facility

The NASA Data Processing Facility (NDPF) is a job oriented facility. It will provide image processing support to EOS similar to that provided the ERTS project. The NDPF is designed to produce precision quality data primarily in digital form to the designated user agencies and selected investigators.

The facility composition is that of an Imager Processing Area and a Data Services Laboratory. The Image Processing Area will process instrument video data. These data will be input in magnetic tape form from all sites other than NTTF where direct data handling is accomplished.

The Data Services Laboratory will organize the production scheduling of all data processing, perform data accounting and serve as the interface with the EOS Data Investigations for processed data. The NDPF includes a storage and retrieval system for all data and provides for user services as required by the Data Investigations Program. The NASA Data Processing Facility (NDPF) at GSFC is one of the facilities designed to receive the instrument data in raw form and perform metric correction and conversion to required data types such as film. It is anticipated that regional readout and analysis stations with similar capability will be economically feasible and will exist in the EOS time frame. Also available will be a low cost Local User Terminal (LUT). Spacecraft data system is designed to support the LUT as well as regional ground data handling facilities.

3.2.3.4 Low Cost Ground Stations

In addition to transmission of sensor data to NASA ground stations, the EOS will be capable of transmitting data directly to a number of small local user stations (see Section 3.2.1.8). These low cost stations will provide users with a direct data source for information which is of immediate importance to them, eliminating the dependence of the local user upon a central data collection and processing facility.

These stations will in general have a limited capability in both processing and data rate reception. The limited geographic interest of the local user results in reduced data quantity. Reduction in data transmission rate is therefore possible while retaining the same data quality or resolution as obtainable at the central facility.

Low cost ground station equipment would typically include a small antenna, receiver, data synchronization and detection equipment, data processing (mini-computer) and suitable display equipment.

3.2.4 Launch Vehicles

As described in Section 3.2, the modular spacecraft can be launched on either the Delta or the Titan vehicles as well as the shuttle depending on payload configuration, weights, and desired orbit altitude. During the system definition studies, trade-off will be performed for payload requirements, and program costs versus launch vehicle selection.

Optimization of NASA funding requirements is best achieved by flying groups of instruments with a few large launch vehicles. Furthermore, utilizing the relaxation of physical constraints to reduce costs and to permit in orbit resupply becomes economically very attractive.

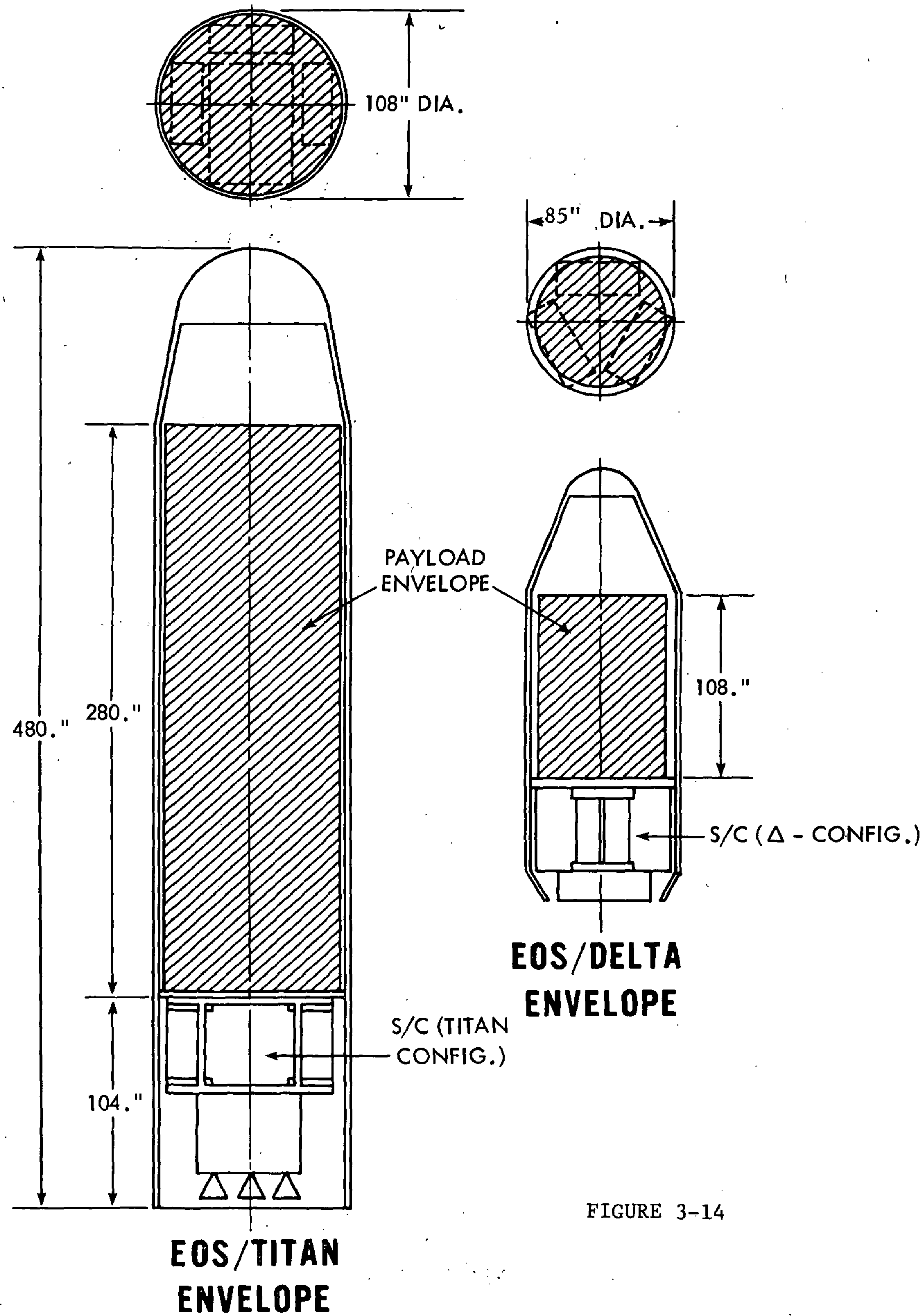
However eventual operational use of EOS components must allow user agencies the flexibility of also optimizing their budgets. Therefore it is vital to keep a smaller EOS a viable option. The launch vehicle for such a S/C would be the present Delta 2910.

The logical candidate launch vehicle for the pre-shuttle launches are:

1. Delta 2910.

2. Titan IID with integral S/C propulsion.

The Delta 2910 is suitable for the smaller volume, non-resuppliable missions flying single instruments. But for a fully shuttle-compatible EOS configuration the Titan class of launch vehicles must be used. Payload clearance envelopes are shown in Figure 3014.



108" DIA.
85" DIA.
480."
280."
104."
PAYLOAD ENVELOPE
108."
S/C (Δ - CONFIG.)
S/C (TITAN CONFIG.)
EOS/DELTA ENVELOPE
EOS/TITAN ENVELOPE

FIGURE 3-14

As Figure 3-15 indicates, the performance characteristics of the Titan IIID/NUS more nearly matches the performance of the Space Shuttle. Therefore it presents the same capabilities and problems as the shuttle.

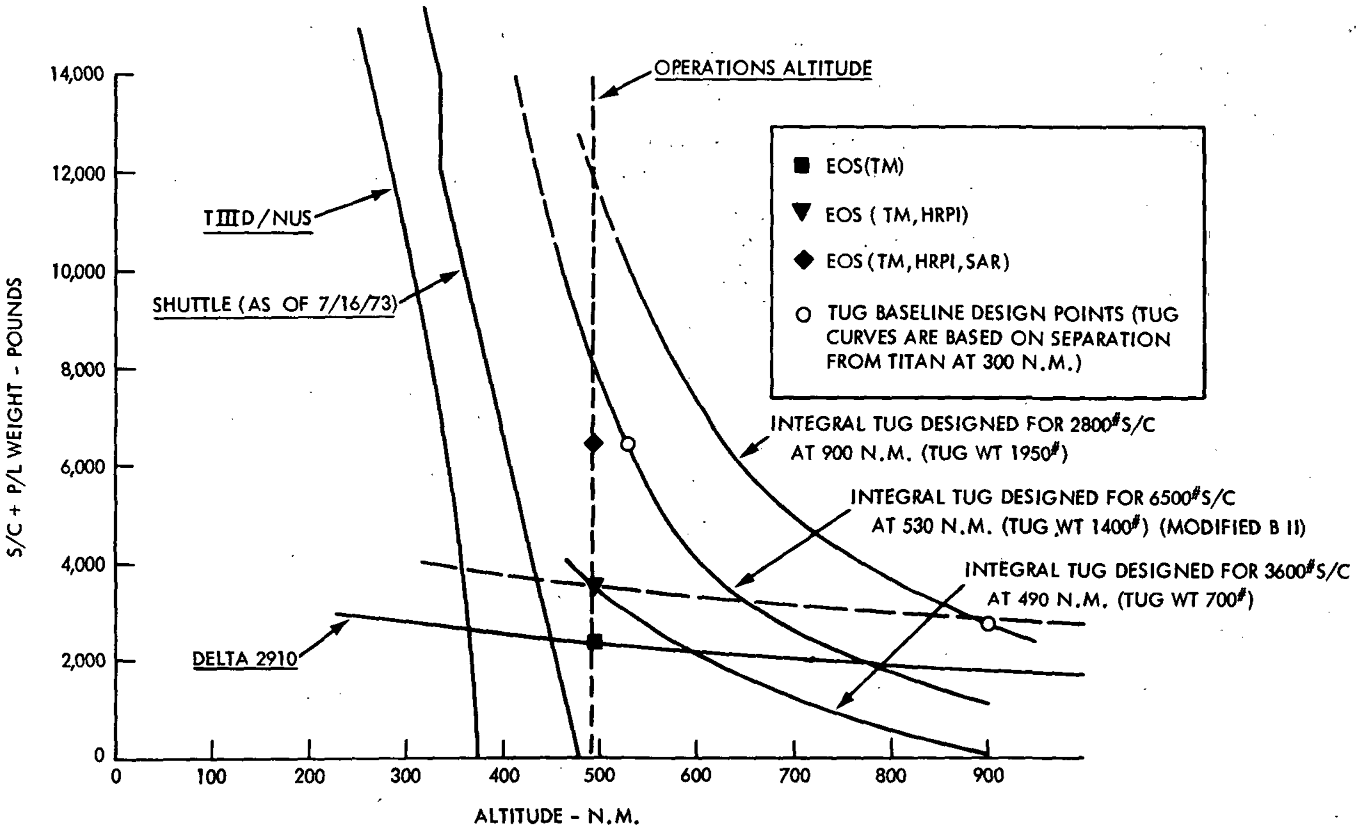
From the standpoint of cost, the Delta 2910 takes four flights to carry all instruments and the Titan IIID requires only one launch. If the only instruments are the Thematic Mapper and the HRPI, then it may be feasible to carry both on one flight by either the Titan or Delta.

In place of an upper stage for the Titan IIID, a cluster of four small solid rockets is added to the orbit adjust package of the EOS. Thus, the orbit adjust package serves as an integral tug to place the EOS at the required operational altitude, to correct for minor injection errors, to perform altitude trim periodically as required for operations and finally to return the EOS to a shuttle orbit for resupply should the EOS operational altitude be higher than Shuttle capabilities.

The incorporation of this tug function into the pneumatics/orbit adjust package not only eliminates the necessity for an upper stage for the Titan IIID, but also provides insurance against any possible shuttle performance degradation during its development cycle. In this way no compromise in EOS operational altitude need to be made to facilitate eventual shuttle launch, resupply and retrieval.

3.2.5 Resupply Systems

The baseline EOS is compatible with the shuttle's launch, resupply and retrieval capabilities. The interface between the standard shuttle and mission peculiar spacecraft is called the Resupply System. Besides the standard shuttle which includes one SAMS boom, T.V. system, lighting system and display system, the Resupply System consists of four basic elements (Figure 3-16A).



OPERATIONS ALTITUDE
EOS(TM)
EOS (TM, HRPI)
EOS (TM, HRPI, SAR)
TUG BASELINE DESIGN POINTS (TUG CURVES ARE BASED ON SEPARATION FROM TITAN AT 300 N.M.)
TIIID/NUS
SHUTTLE (AS OF 7/16/73)
DELTA 2910
INTEGRAL TUG DESIGNED FOR 2800#S/C AT 900 N.M. (TUG WT 1950#)
INTEGRAL TUG DESIGNED FOR 6500#S/C AT 530 N.M. (TUG WT 1400#) (MODIFIED B II)
INTEGRAL TUG DESIGNED FOR 3600#S/C AT 490 N.M. (TUG WT 700#)
S/C + P/L WEIGHT - POUNDS
14,000
12,000
10,000
8,000
6,000
4,000
2,000
0
100
200
300
400
500
600
700
800
900
ALTITUDE - N.M.

FIGURE 3-15

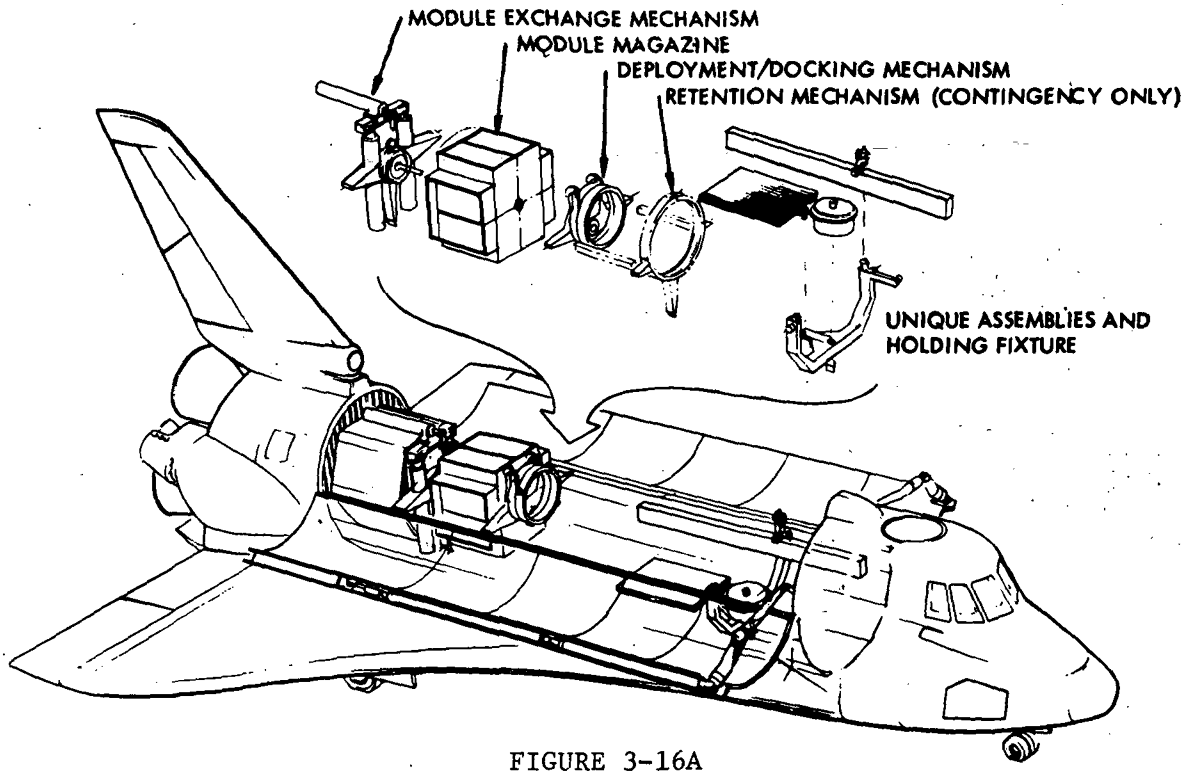


FIGURE 3-16A

EOS FLIGHT SUPPORT SYSTEM INSTALLATION REFURBISH MISSION.

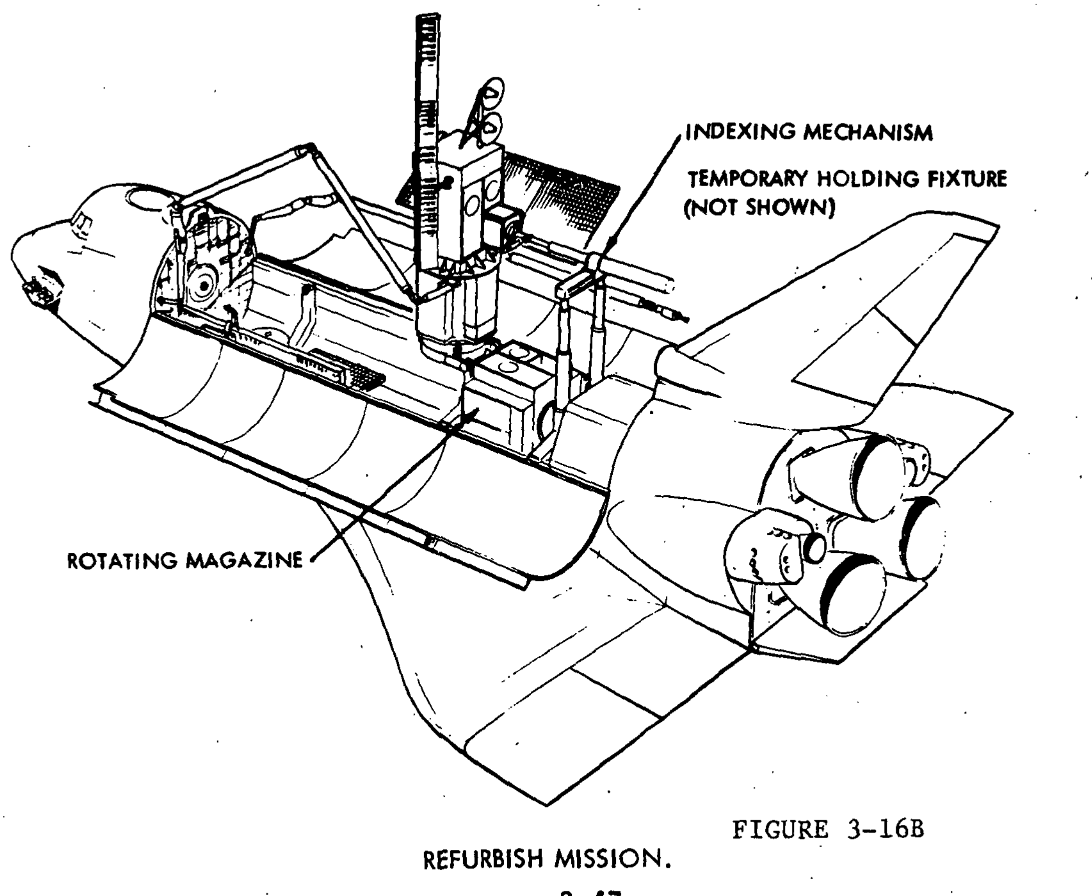


FIGURE 3-16B

REFURBISH MISSION.

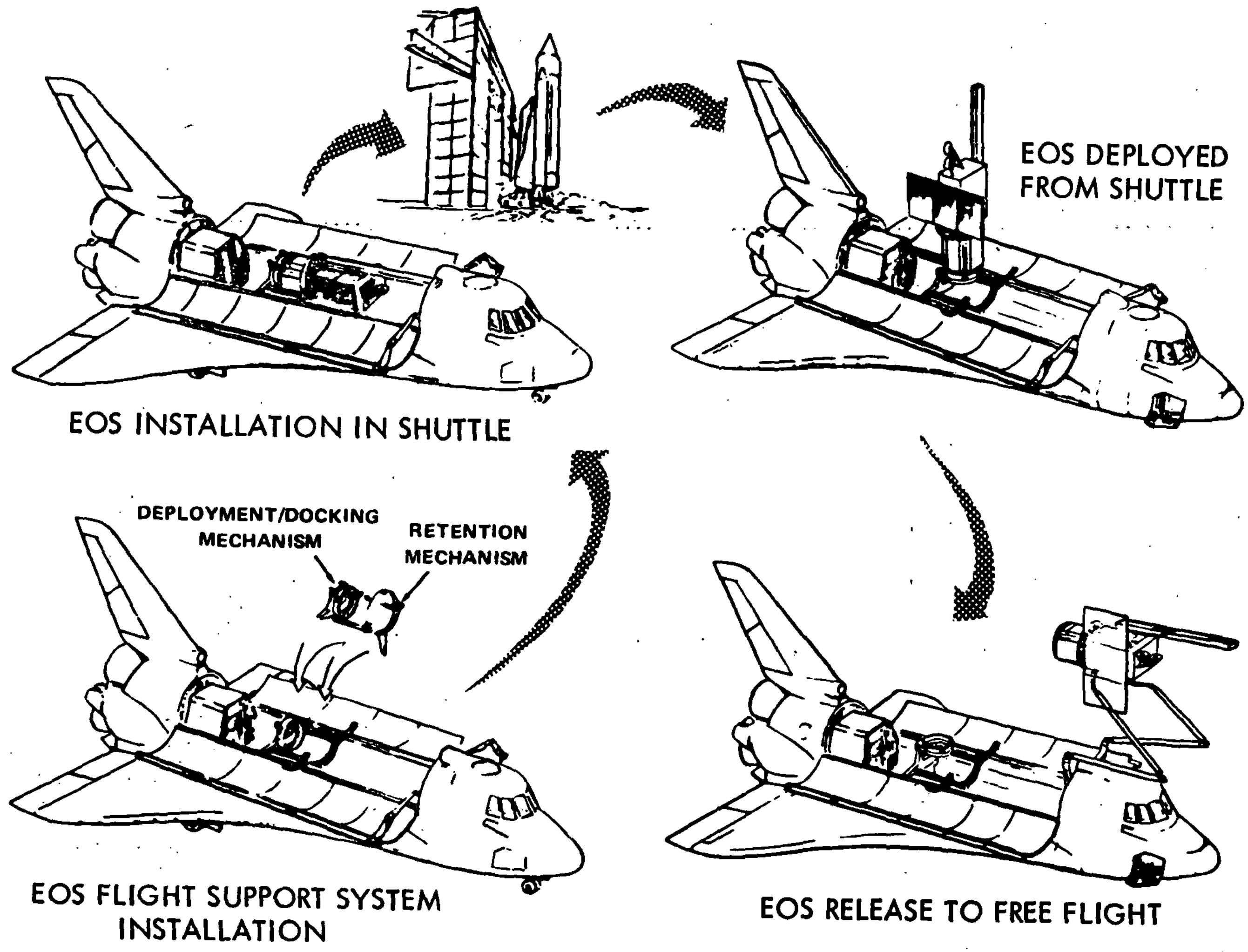
A. FLIGHT SUPPORT SYSTEM (FSS)

1. A retention ring (tie down) structural assembly.
2. A rotating docking platform.

B. RESUPPLY MANIPULATOR SYSTEM

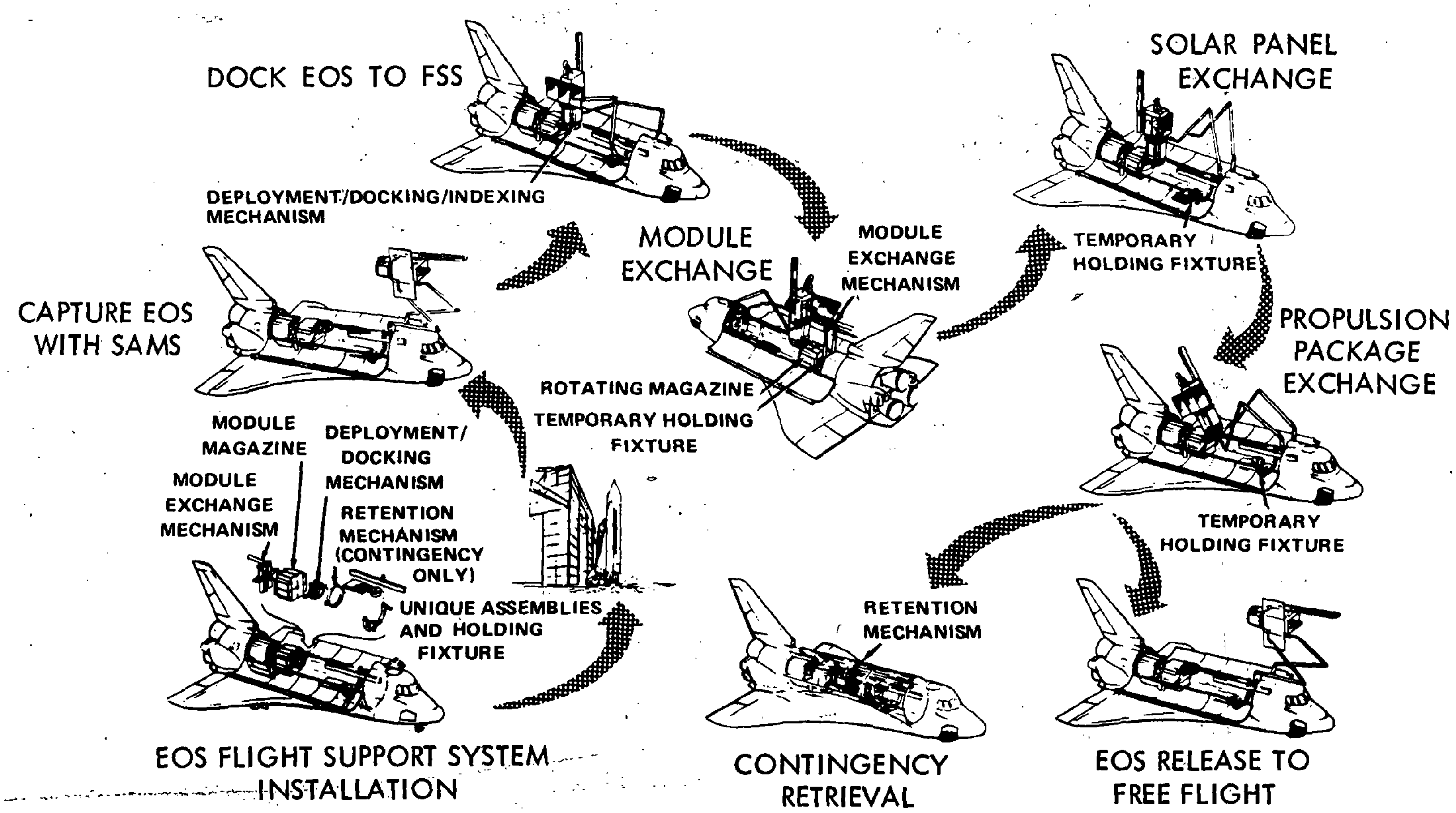
1. A specialized resupply manipulator assembly.
2. A module storage assembly.

In the launch mode the observatory is cradled in the retention ring assembly. In this mode the only structural contact that the observatory has with the retention assembly is at the transition ring. Thus the spacecraft design can be approached with complete independence relative to the shuttle's structural flexibilities, and by the same token the shuttle is not influenced by the spacecraft rigidity characteristics. Once in orbit the tie down clamp of the retention ring is released and spacecraft is free to rotate up vertically from within the bay (Figure 3-17 and 3-18).



EOS LAUNCH MISSION SEQUENCE.

FIGURE 3-17



EOS REFURBISH MISSION SEQUENCE.

FIGURE 3-18

The docking platform is the mechanism which rotates the spacecraft out of the shuttle cargo bay to the 90° erect position. Just prior to the release of the trunion clamp, a set of dog latches are engaged which clamps the lower deck of the spacecraft to the platform. Then after release of the clamp band, the platform is rotated 90° thus elevating the spacecraft. Once the spacecraft has been checked out, it can be released by actuating the latches to the release configuration.

In the resupply mode the EOS is approached by the shuttle to within 30 feet. At this point the shuttle attached manipulator boom (SAMS) is used to link the observatory to the shuttle. The capture is made by the end effector of the SAMS at the transition ring. Under control of the SAMS the ΔV is brought to zero and the EOS is docked to the docking platform. Once docked the latches are engaged and the special purpose manipulator is used first to remove and store the degraded subsystem module and then to replace it with a new one which had been stored in the module storage magazine. To facilitate the resupply activity, both the storage magazine and the docking platform have the capability of rotating 360° in 90° increments. Replacement of an obsolete instrument is shown in Figure 3-16B.

Either planned or emergency retrieval is accomplished by capture with the SAMS, then by docking and rotating the spacecraft down onto the retention assembly. Once this is accomplished, the clamp band is closed around the transition ring and the latches on the docking platform are released. The EOS is now ready for return to earth.

3.3 MISSION AND DATA OPERATIONS

3.3.1 STDN SUPPORT

Network support for EOS includes three STDN sites for the Terrestrial Survey mission. The stations are Goldstone, California; Fairbanks, Alaska; and NTTF (GSFC). A fourth station (Madrid) is required for the ocean/met mission but could be eliminated with the availability of the Tracking & Data Relay Satellite (TDRS). Table 3-C describes briefly the STDN support requirements for the first EOS mission.

Goldstone passes will provide coverage of the western and west-central portions of the continental United States. Fairbanks will provide coverage of Alaska. Support requirements of both sites are identical; to receive and record instrument video data, receive S/C housekeeping telemetry data, and transmit OCC generated commands to the EOS. Video data tapes from both sites will be shipped to GSFC for processing.

The Network Test and Training Facility (NTTF) at GSFC will provide real time coverage of the central and eastern United States. NTTF will provide real time instrument video data to the OCC and NDPF. S-Band tracking, S/C telemetry and direct real time command capability will be provided.

Improvements or modifications to the present network capability are required to support the EOS missions. Receivers, bit synchronizers, and tape recorders will be required to support the greater than 100 megabit per second data rates from the imaging instruments. The instrument video data will be transmitted over an X-band link. This link can be supported by a six foot ground antenna as an alternative to modifying existing antenna systems.

TABLE 3-C

DATA ACQUISITION AND SUPPORT STATIONS FOR EOS-A

STATION	OPERATION
Goldstone, California	S-Band Command, Tracking and Receive Telemetry.
	Receive and Record S-Band Video
Fairbanks, Alaska	S-Band Command, Tracking and Receive Telemetry.
	Receive and Record X-Band Video
Network Test and Training Facility	S-Band Command, Tracking and Receive Telemetry.
	Receive X-Band Video

3.3.2 NASCOM SUPPORT

The Spaceflight Tracking and Data Network (STDN) is supported by NASCOM which will supply the communication links between the STDN and the EOSOCC at GSFC. Basic communication requirements will include voice, data and command links to all sites to support real time operation of the EOS. A video link for instrument data will be required between NTTF and the OCC as additional support. All other links are standard. Detailed support requirements are described in the EOS Support Instrumentation Requirements Document (SIRD).

3.3.3 OCC/NDPF OPERATIONS AND MAINTENANCE

The Operations and Maintenance (O & M) of the EOSOCC and the NDPF is the responsibility of the Mission and Data Operations Directorate. This responsibility includes coordination of OCC, network and NDPF activities in supporting the EOS Project mission operations requirements. The production of data products and delivery to the users, preparation of data catalogs and the maintenance of a data archive are additional responsibilities.

Funding for contractor manpower for the M & O of the facilities, supplies, etc. necessary to support pre and post launch phases of EOS will be provided by OTDA.

3.3.4 ORBITAL COMPUTATION

The Operations Support Computing Division and Mission Support Computing and Analysis Division will provide orbital computation support for the EOS missions. This will include mission planning analyses during the EOS study phases, pre-launch simulations, real-time launch support, and routine post-launch operational support. Tracking data from the network will be processed for predicted and definitive spacecraft ephemerides, orbit adjust calculations, recommendations, and post-execution evaluations; and prediction of station acquisition and timelines necessary for operations planning and network scheduling.

3.4 LOGISTICS AND FACILITIES

There are no requirements for special transportation systems or vehicles to handle the EOS. Its width (less than 10 feet in diameter) allows shipment by conventional means. The length which is a direct function of instrument length, also does not present a transportation problem since the observatory is designed for horizontal loading into the space shuttle. Therefore it can be shipped in a horizontal position.

Shuttle cargo loading equipment will be required to place the EOS into the cargo bay consistent with payload cleanliness requirements. At present this type of equipment will be provided by the shuttle program as launch site support equipment.

Again, because of mechanical design of the EOS, no new facilities for fabrication and/or test are required. Emphasis will also be placed on utilizing and/or modifying existing NASA GSE were applicable as opposed to development of new GSE.

The individual modules on EOS are being designed to be insensitive to their placement in the system. In this manner the subsystem and instrument thermal vacuum tests will be as realistic as the system level test. Therefore, system level thermal vacuum testing is not mandatory.

In order to assure that the systems contractor emphasizes this design objective, systems level thermal vacuum testing will not be allowed at his plant. The first EOS will be brought to GSFC for such a test. If completed without significant difficulty, subsequent missions may omit this test.

EOS FACILITY REQUIREMENTS

DESCRIPTION	LOCATION	DATE REQUIRED	DURATION
1. Medium Sized Thermal Vacuum and Vibration Test Facilities	Subsystem and Instrument Contractor Plants	TBD	TBD
2. Large Scale Thermal Vacuum and Vibration Test Facilities	GSFC	1978	6 Weeks
3. Pre-launch Checkout Area	Space and Missile Test Center	4 Weeks prior to launch	Thru Launch
4. Launch Site	Space and Missile Test Center	4 Weeks prior to launch	Thru Launch

3.5 RELIABILITY, QUALITY ASSURANCE AND TESTING

The requirements for the R&QA and test program will be subject to a critical review during the system definition phase. The R&QA program will be responsive to the intent of applicable portions of the NASA R&QA publications; however, some requirements and documentation that are normally invoked may be omitted if risk assessment and cost effective judgement so dictates. The proposer will be given the latitude to provide alternative approaches to obtain the R&QA and test goals. Each element of the R&QA and test program will be evaluated for effectiveness.

The test program concept will be to review environmental testing at all levels to ascertain the necessity of each test and minimize redundancy and overtesting. Where possible, previously designed systems of proven reliability will be utilized to reduce qualification level testing. Cost can be substantially reduced by performing acceptance environmental tests at the subsystem level after components have been qualified. Omission of the component level environmental test will reduce the total number of tests significantly. Technical risks are not incurred since all equipments will still be exposed to appropriate environments at the subsystem level.

The initial system level thermal-vacuum test will be performed at GSFC. It is expected that subsequent missions will not require such a test.

Simplification of R&QA requirements is planned. The System Definition competitors will propose which requirements to retain and which to eliminate. The System Execution RFP will be synthesized from all recommendations and a common set of requirements specified. Those NASA requirements normally used but not included under the reduced set specified will be priced as options. NASA Management may then decide on their inclusion or omission.

3.6 ADVANCED TECHNOLOGY REQUIREMENTS

EOS missions do not appear to require any major technological breakthroughs, but in some areas extrapolations within the state-of-the-art will be necessary. Such extrapolations yet to be completed are:

SAR: Improved TWT cathode life under high peak-power operation; state-of-the-art life is probably adequate, but more margin would be desirable.

TM: Probably none.

HRPI: Calibration methodology; will be studied during System Definition results should enable systems contractor(s) to incorporate during System Execution.

DCS: None.

PMMR: Probably none.

PMI: Choice of adequate measuring techniques; measurement of type and quantity of aerosols is extremely difficult with any currently proposed technique; supportive R&D in this area is continuing at LRC.

Data Management:

a. Improved resolution capability for ground equipment producing multi-color imagery.

b. All digital image correlation and change detection.

c. Responsive interactive data processing capabilities for users.

Various Government and industry groups are pursuing techniques in these areas. The results to date are encouraging; cost-effective solutions are expected in the time frame needed for EOS.

Data Handling:

a. 200 MHz channel width X-band receivers.

b. 200 megabit per second bit synchronizers.

c. 200 megabit per second tape recorder (for ground use).

Government studies and industry efforts have demonstrated the feasibility of 200 megabit per second digital data handling technology. While equipment does not exist today, breadboards of high speed data circuits and tape recorder techniques have operated at the required data rates envisioned for EOS.

In summary, technological problems that do exist should be overcome prior to release of the RFP for System Execution.

3.7 ANALYSIS OF MISSION RESULTS

The application of the data from the instruments of the Land Resource Management. Instrument Development Mission may be subdivided into several phases.

1. Instrument Validation

2. Data Analysis

3. Application Development

1. Instrument Validation - In this initial phase of the flight mission instrument data are analyzed by the Facility Instrument Teams to establish that the instrument is working properly, that the calibration factors are being properly applied, to develop any corrections required for atmospheric effects, to compare with appropriate surface truth or other correlative information, and to develop initial algorithms for data analysis.

2. Data Analysis - In parallel with the data validation phase, processed data are released to users for experimental application in the discipline areas of their concern. It is expected that users will be of several types paralleling the user groups developed in the ERTS Program. As a result of announcements of data application opportunity, a limited number of experimenters will be funded by NASA for investigations that evaluate discipline contributions resulting from higher resolution, or data in new or different spectral bands, as compared to the prior ERTS data or concurrent data obtained in the Earth Resource Operational System.

3. Application Development - The principal U.S. government agencies concerned with Land Resources are USDI, USDC, USDA, USACE. In parallel with the instrument validation and data analysis phases, it is expected that these agencies, especially their research organizations, will be involved in a number of experimental applications of the data to their resource inventory and management problems. This application development phase will make use of the regional readout and analysis stations, and the real time availability of selected data to support such activities as resource modeling, and trade-offs between land use inventory by national sample (Thematic Mapper) and by statistical sampling (HRPI).

Pollution, Oceanography, Weather and Climate Data. For these data, much the same phases as the Land Resource Management data will apply. Since these data are global, the principal users are the national services such as NOAA, the USN Fleet Weather Control, the National Marine Fisheries Service. To support these users and other experiments effectively, preliminary data processing to produce computer compatible data will take place at the NASA Data Processing Facility (NDPF). Since many of the user needs require not the measured radiance data but such extracted quantities as ocean color, ocean temperature, or pullution distribution, it is probable that data processing to the map stage would be most cost effective, and permit distribution of such extracted data at a lower data rate and lower cost than would be possible if radiance data were to be distributed.

3.8 ANALYSIS OF ENVIRONMENTAL IMPACT

In accordance with the National Environmental Policy Act (NEPA) of 1969, the OSS Director of Launch Vehicles and Propulsion Programs has issued a draft Environmental Statement, dated August 1, 1972, describing the known and anticipated environmental effect of the various launch vehicles used by NASA. After coordination of Government-wide comments and revisions, the statement will be formally promulgated by NASA as prescribed by the NEPA.

The EOS mission will be launched on vehicles, the effects of which are described in detail in the draft statement. The environmental impact of this launch, even in the event of failure or abort, is not considered to be significant in that the effects are limited in extent, duration, and intensity.